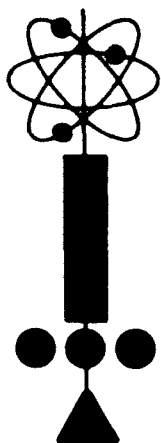


CONTRACT NO. NAS 8-11207

GE NO. ~~66~~SD2019



ELECTRICALLY-PROPELLED CARGO VEHICLE FOR SUSTAINED LUNAR SUPPLY OPERATIONS

SUMMARY REPORT

PREPARED FOR:
GEORGE C. MARSHALL SPACE FLIGHT CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
HUNTSVILLE, ALABAMA 35812

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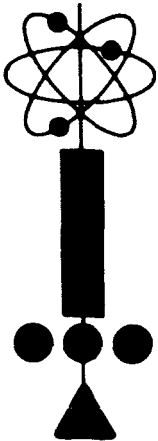
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MISSILE AND SPACE DIVISION
ADVANCED NUCLEAR SYSTEMS ENGINEERING

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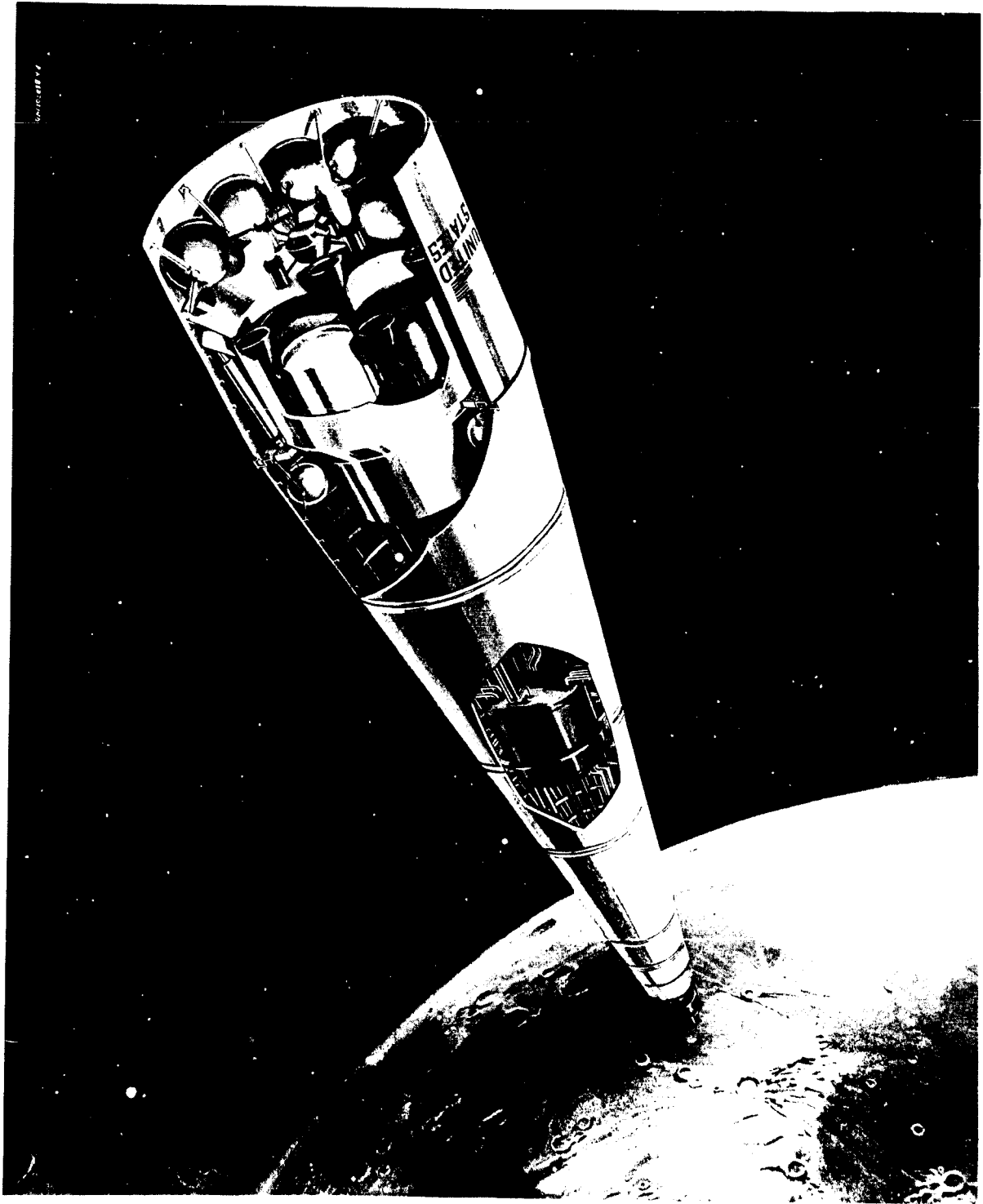
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GLOSSARY

Booster Cost	- cost to procure and launch a two-stage Saturn V rocket vehicle
Cargo Delivery Cost	- cost to deliver a unit mass of cargo from the earth to the lunar surface
Cargo Fraction	- Vehicle Cargo divided by gross mass of lunar landing vehicle
Cargo Requirement	- cumulative or total cargo to be delivered to the lunar surface over the service life of the powerplant concept
Cost Index	- ratio of electrically propelled vehicle system to reference chemical rocket vehicle system cost per unit mass of cargo delivered
Development Cost	- non-recurring cost for research, development, test and evaluation of powerplant and electrical propulsion system
Development Cost Index	- Development Cost divided by Booster Cost
Nuclear Powerplant Cost Index	- cost per metric ton of powerplant divided by Booster Cost
Power	- net electrical power output of powerplant and input to electrical propulsion system
Powerplant Cost	- recurring cost for procurement of one operationally qualified powerplant
Powerplant Specific Mass	- ratio of powerplant mass to net electrical power output
Propulsion Time	- time duration of electrical propulsion during earth-orbit to lunar orbit transfer
Time Worth Index	- fractional increase in apparent cost per year of trip time
Trip Time	- time duration for transfer from earth-orbit to lunar orbit
Vehicle Cargo	- cargo delivered per trip per electrically propelled system

SECTION I
INTRODUCTION



SECTION 1

INTRODUCTION

The General Electric Company, under contract to the NASA George C. Marshall Space Flight Center, has performed a study to determine the applicability and the operating modes of electrically propelled earth-moon shuttle vehicles for logistic support of advanced lunar operations. The program was started 3 June 1964 and has run 24 months. This document is the final summary report. Two documents were previously issued under this contract. They are:

- Document GE No. 65SD4361, "Electrically-Propelled Cargo Vehicle for Sustained Lunar Supply Operations, Final Report," 28 June 1965
- Document GE No. 66SD5200A, "Electrically-Propelled Cargo Vehicle for Sustained Lunar Supply Operation," Nuclear Powerplant Analysis Topical Report, 27 May 1966

The electrical propulsion systems presently envisioned provide a low level of thrust, which restricts its application to propulsion of vehicles already in self-sustaining orbits. Thus, the operational pattern for logistic material support of lunar operations is performed in three stages:

- 1) ascent from earth to orbit by use of high thrust chemical rockets,
- 2) transfer from earth to lunar orbit by use of electrical propulsion, and
- 3) descent from orbit to the lunar surface by use of high thrust chemical rockets.

Each of the three travel stages can be performed in a variety of ways, such as single-use versus reuseable vehicles. However, this study has been confined to the stage for transfer from earth orbit to lunar orbit. The ascent from earth to orbit is assumed to

be accomplished by use of the Saturn V; the descent from orbit to the lunar surface, by a single-use, oxygen-hydrogen propelled vehicle scaled to optimum size.

The reason for considering electrical propulsion for logistic material support of lunar operations is that by this means substantially higher specific impulses (2000 to 10,000 seconds) can be achieved than by chemical (340 seconds) or nuclear rockets (850 seconds). The higher specific impulse reduces the demand for propellant mass and the mass savings can be translated into more payload or lunar cargo. The all-chemical Saturn V system can place a 109 metric ton net payload into earth orbit, and a 31.7 metric ton net payload into lunar orbit. The mass requirement to accomplish the orbit transfer is 77.3 metric tons. The complete elimination of this orbit transfer mass requirement would increase the lunar orbited payload to 109 tons, which is an increase of 243 percent. The electrical propulsion studies have indicated that half of this theoretical increase can be achieved in practice. The cargo increase can yield a corresponding cost savings for the logistics operation, which is reduced by the cost to develop and construct the powerplant and electrical propulsion system. This cost consideration has been factored into the analysis and has been found to be very significant.

The program was conducted in four stages. The first stage was directed at performing a generalized mission analysis using nuclear power sources for the electrical propulsion system, wherein the nuclear powerplant characteristics were represented in parametric form. This study was conducted for thirteen months, and the results were presented in GE Document No. 65SD4461. Next, a detailed study was performed to estimate the mass and size versus power characteristics of eight assumed powerplant concepts. These concepts were based on postulated "Early," "Intermediate," and "Advanced" technology levels of thermionic, Rankine, and Brayton systems. This study was conducted for six months, and the results were presented in GE Document No. 66SD5200A. The third stage was directed at solar power sources for electrical propulsion, which involved a detailed trajectory analysis to determine the differences in propulsion requirement and trip time between nuclear systems with continuous power and solar systems with discontinuous power due to the earth shadow effect. The results of this effort are presented in the

Appendix to this report. Finally, a mission study was performed for evaluation of the particular nuclear and solar power systems previously identified for which mass versus power relationships had been determined. The results of all these studies are summarized herein. The final summary report has been prepared to cover all aspects of the electrical propulsion operation, omitting details covered in previous reports.

The contract study was performed by the Advanced Nuclear Systems Engineering Section of the GE Missile and Space Division at King of Prussia, Pennsylvania. Assistance was provided by other General Electric divisions, departments and sections as appropriate in specialized areas.

SECTION II

SUMMARY

SECTION 2

SUMMARY

The study of electrically propelled earth-moon shuttle vehicles for logistic support of advanced lunar operations has involved many considerations. A substantial number of variations of the operational mode have been identified and compared. The propulsion requirements have been determined for both continuous propulsion (characteristic for nuclear power sources) and discontinuous propulsion (characteristic for solar power sources). The component parts of the electrically propelled spacecraft have been examined, with particular attention directed at nuclear powerplant characteristics. These operating mode and spacecraft characteristics have been used to formulate mathematical models of the lunar cargo operation, and the optimum mission modes and spacecraft parameters have been determined. This study has required the use of many judgements or assumptions, which can not be substantiated within the scope of presently available data. However, these have been used primarily for comparison between systems of different types, wherein the objective is to bound the problem and provide consistency for evaluation.

The use of electrical propulsion can show a substantial payload advantage, which is approximately 150 percent greater than that attainable by the alternative of chemical rocket propulsion. However, the cost of the powerplant and electrical propulsion system is quite significant and needs to be factored into the mission analysis for optimization of spacecraft parameters. The inclusion of this cost into the analysis tends to increase the system cost, including booster, by about 20 percent, which can still yield a 50 percent cost reduction favoring electrical propulsion. The increased payload must be sufficient to more than compensate for the cost of the powerplant and propulsion system. The mission study has been based on the consideration of both non-recurring costs (RDT and E program) and recurring costs (powerplant and

propulsion system procurement). The inclusion of non-recurring costs also requires that the cumulative lunar cargo requirement be factored into the study. Comparisons made between nuclear and solar power sources using the above approach have tended to favor the nuclear systems.

Many operational modes have been determined, but the principle types are the following:

- Single-trip operation
- Single-powerplant shuttle operation
- Multiple-powerplant shuttle operation

Variations of these types are based on number of boosters used to initiate the operation, staging and replacement of electric thrusters, and disposal of spent propellant tanks. A prime consideration in the case of nuclear powerplants is that the number of trips to be undertaken is between one and four because of the long trip times anticipated. As a result the placement of the nuclear powerplant in orbit has to be considered in the mission analysis. The conclusion determined from the mission study was that the single-trip operation was the best mode when powerplant life was a limitation having a pronounced effect on the cost of the RDT and E program. It appeared that the powerplant life and reliability could be improved after starting operational use in a single-trip mode so that eventually a shuttle operation could be developed. Single-trip operation was also favored by considerations in addition to performance such as elimination of earth rendezvous requirement.

The solar power approach, on the other hand, favored shuttle mode operation. The reason here is that the initial cost of a solar array would probably exceed the Saturn V booster cost. Therefore, the solar array would have to be amortized over many missions to be cost effective. However, solar cells are inherently long life and five years operation in space has already been demonstrated. Thus, the mission analysis for solar systems was based on the shuttle mode of operation.

The electrically propelled space vehicle is comprised of the following parts: lunar landing vehicle, electric thrusters, propellant system and electrical power supply. The lunar landing vehicle was assumed to employ oxygen-hydrogen rockets for propulsion, which was estimated to provide the lander with a capability of 0.40 to 0.45 cargo fraction. This cargo fraction was substantiated by preliminary analysis. The long trip times was considered in the analysis of the cryogenic storage, and the effect was not very significant because of the rather large size tanks resulting in the electrical propulsion system large payload capability. The electric thrusters were based on the electron-bombardment ion jet characteristics. Comparisons were made between various thrusters which provided substantiation to this selection. Propellant system mass was based on a tank plus structure assumed at 10 percent of the propellant mass.

The nuclear powerplant study encompassed a preliminary mass and size estimation for eight powerplant concepts. These concepts include three levels of technology assumed for Rankine and Brayton cycle systems, and two levels of technology assumed for thermionic systems. The specific mass was determined for a range of power and subsequently curve-fitted for mission evaluation. A typical comparison of the eight powerplant is presented in Table 2-1, wherein the powerplant mass is set at 11,300 kg (25,000 lb). Comparison at equal mass is more appropriate for mission evaluation because the optimum powerplant mass variation between various systems is small compared to the power variation. This comparison in Table 2-1 shows power varying between 0.84 and 3.13 megawatt for the various systems. The thermionic and Rankine systems are close in power level, and the Brayton cycle system tends to be lower by a factor of 2 to 3.

TABLE 2-1. NET POWER OUTPUT OF VARIOUS NUCLEAR POWERPLANTS SIZED AT 11,300 KG TOTAL MASS

SYSTEM	TECHNOLOGY LEVEL		
	Advanced	Intermediate	Early
Thermionic	3.13	1.93	-
Rankine	2.94	1.95	1.15
Brayton	1.17	.99	.84

The solar powerplant is estimated to have a potential of 10 KG/KW_e, but 20 KG/KW_e is about the best for which any panel sections have been actually fabricated. The cost is the most significant item, and this is presently projected at \$500 per watt. On this basis, the cost of one megawatt array would be \$500 million.

A mission comparison has been performed between the various nuclear and solar powerplants on a consistent set of assumptions, and the results are presented in Table 2-2. In this comparison, the powerplant electrical output and thruster specific impulse have been optimized on the basis of cost wherein a cost penalty due to trip time is applied. This cost penalty is applied by multiplying the Cost Index by the factor, $(1 + 0.5 T)$, where T is the trip time in years. This somewhat arbitrary approach was found to be quite effective for determining particular design points for each of the powerplant concepts to use in comparisons. Other assumptions used in preparing Table 2-2 include \$100 million assumed booster cost, \$2000/kg nuclear powerplant cost, and 2000 metric ton cumulative lunar cargo requirement.

The comparison of powerplants presented in Table 2-2 indicates many interesting conclusions and is dependent on many qualifications. Comments pertinent to these indications are as follows:

- Lunar landing vehicle gross mass -- falls in narrow range of 75 to 82 metric tons; thus, its size can be selected independent of electrical power supply.
- Nuclear powerplant gross mass -- falls in narrow range of 7.6 to 12 metric tons.
- Power and specific impulse -- very dependent on power supply selection.
- Cost index -- also falls in narrow range showing 50 percent cost advantage for most systems.

TABLE 2-2. ELECTRICALLY-PROPELLED LUNAR CARGO
VEHICLE PERFORMANCE CHARACTERISTICS SUMMARY

Type Technology Mode	T A S	T I S	R A S	R I S	R E S	B A S	B I S	B E S	S _O A M	S _O A M	S _O E M
Power, MW _e	2.20	1.54	1.85	1.41	.96	1.23	1.02	.90	1.87	1.52	1.31
Specific Impulse, sec	6372	5413	5891	5209	4416	4906	4537	4270	5132	4295	4907
Trip Time, hr	3720	4695	4169	4961	6444	5453	6177	6738	5264	5553	7699
Cost Index	.4834	.5041	.4722	.4939	.5287	.5514	.5598	.5746	.5049	.5987	.5305
Specific Mass, KG/KW _e	3.95	6.08	4.13	6.00	9.90	9.48	11.27	13.22	10.00	10.00	20.00
Mass Summary, metric tons											
Powerplant	8.69	9.36	7.63	8.46	9.50	11.66	11.49	11.91	18.70	15.20	26.20
Thrusters	3.70	2.70	3.16	2.50	1.80	2.23	1.90	1.71	3.34	2.88	2.37
Propellant System	14.11	16.44	15.21	17.06	19.87	18.01	19.43	20.45	25.33	28.54	29.18
Landing Vehicle	82.50	80.50	83.00	80.98	77.83	77.10	76.18	74.93	80.23	77.58	77.45
TOTAL	109.00	109.00	109.00	109.00	109.00	109.00	109.00	109.00	127.70	124.20	135.20
CARGO	33.00	32.20	33.20	32.39	31.13	30.84	30.47	29.97	32.09	31.03	30.98
Cost Summary, \$10 ⁶											
RDT&E	494	564	491	552	667	682	726	782	—	—	—
Unit	17.38	18.72	15.26	16.92	19.00	23.32	22.98	23.82	374	608	262
<p>LEGEND</p> <p>T = Thermionic System A = Advanced Technology</p> <p>R = Rankine System I = Intermediate Technology</p> <p>B = Brayton System E = Early Technology</p> <p>S_O = Solar System S = Single-Trip Mode</p> <p> M = Multi-Trip Mode</p>											

- Trip time -- very dependent on power supply selection.
- RDT & E cost -- appears to favor advanced systems, but this is result of costing model, which related cost to mass and trip time, independent of powerplant type and technology level. It is conceivable that this approach is approximately correct, but more work is required in this area to improve on the accuracy of the RDT & E cost model.
- Unit cost (procurement of operational powerplant) -- shows little difference between nuclear systems, but a substantial jump for solar systems. Two advanced solar systems are shown, one costed at \$20/watt/year, the other at \$40/watt/year of operation.

SECTION III
MISSION DESCRIPTION

SECTION 3

MISSION DESCRIPTION

Electrical propulsion imparts a thrust force to a free body in space causing an acceleration that is generally in the range of 10^{-4} g. Consequently, the use of electrical propulsion is restricted to use in stable orbits for orbit change and cannot be used in the travel between surface and orbit of either the earth or the moon.

The electrical propulsion system is comprised of an electrical power supply, propellant supply system and electrical thrusters. The electrical power supply can be either nuclear or solar powered. This system and the other components and subsystems are designed for long-time operation. Thus, the electrical propulsion system can function for many thousands of hours, which enables the low acceleration to develop into a substantial velocity increase, or total impulse. It is this capability to achieve a large impulse by integrating the thrust over a long period of time that makes electrical propulsion of interest for space travel. The application of electrical propulsion to the cislunar space transportation system is described below.

The use of electrical propulsion for sustained lunar supply operations is illustrated in Figure 3-1. The travel procedure involves three phases of propulsion. The first phase is the placement of the space vehicle in earth orbit. This is accomplished by use of high thrust chemical rockets, which initiate operation on the earth's surface and boost the space vehicle to a stable orbit around the earth. At this point electrical propulsion can be employed and the second transportation phase commences. The electrical propulsion provides a continually enlarging space vehicle orbit, which eventually breaks into an earth-moon transfer trajectory. After this, the electrical propulsion decelerates the

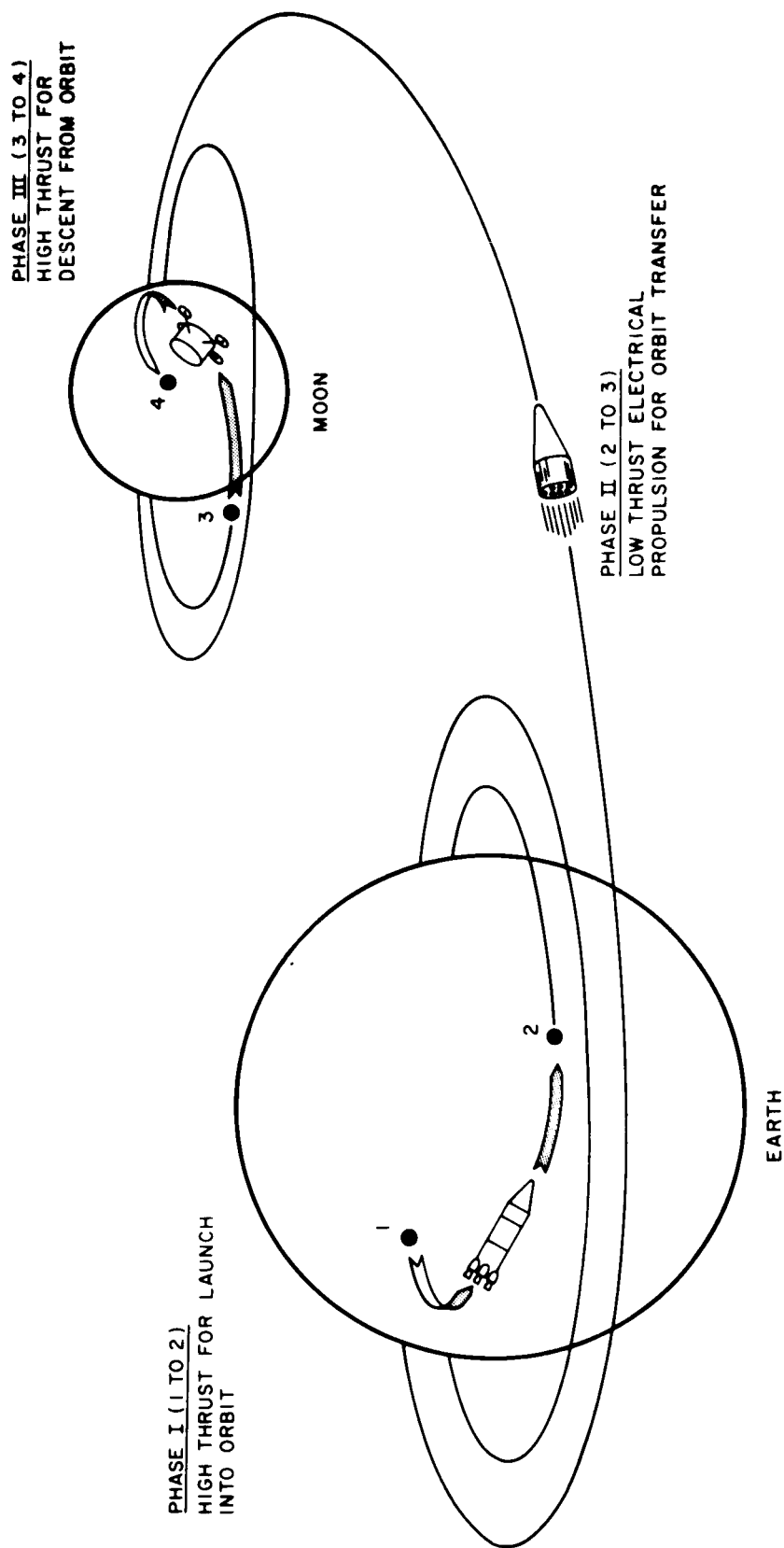


Figure 3-1. Transportation Pattern for Lunar Logistics Using Electrical Propulsion Stage

space vehicle into a low lunar orbit thereupon terminating the second transportation phase. The third phase is the process of transport between the lunar orbit and the lunar surface. This phase requires high thrust propulsion such as chemical rocket and is basically similar to the first phase.

The terminal points for start and completion of electrical propulsion can be varied to suit the equipment selection and mission optimization. In general, the earth orbit terminal point should be selected to be a minimum energy orbit limited only by the atmospheric drag consideration. Similarly, the lunar orbit terminal should be minimal, based on studies performed on mission optimization.

Many operational variations exist for the assembly of an electrically propelled space vehicle. The simplest of these is the single-trip operation mode depicted in Figure 3-2. The earth launch booster considered is comprised of the SIC and SII Saturn stages. The electrically propelled space vehicle is mounted in place of the SIVB stage, and it is comprised of electrical power supply, propellant supply system, electrical thrusters, and lunar landing vehicle including lunar cargo, and propellant for lunar descent. The SIC and SII Saturn stages are jettisoned after burnout placing the electrically propelled space vehicle in a near circular orbit of approximately 560 kilometers altitude. The electrical power supply then initiates operation powering the electrical thrusters and propelling the space vehicle to lunar orbit. At this terminal point the lunar landing vehicle is separated from the electrical propulsion system. The lunar landing vehicle then propels itself from orbit to the lunar surface using high thrust chemical rocket propulsion, and the remaining parts of the space vehicle are abandoned in lunar orbit.

The shuttle mode is an operational scheme for cislunar space transportation whereby the electrical power supply is continually reused. This mode is illustrated in Figure 3-3. The earth launch assembly differs from the single-trip mode by the omission of the electrical power supply. Instead, the earth launch vehicle is required to rendezvous with the electrical power supply in earth orbit. The electrically propelled vehicle

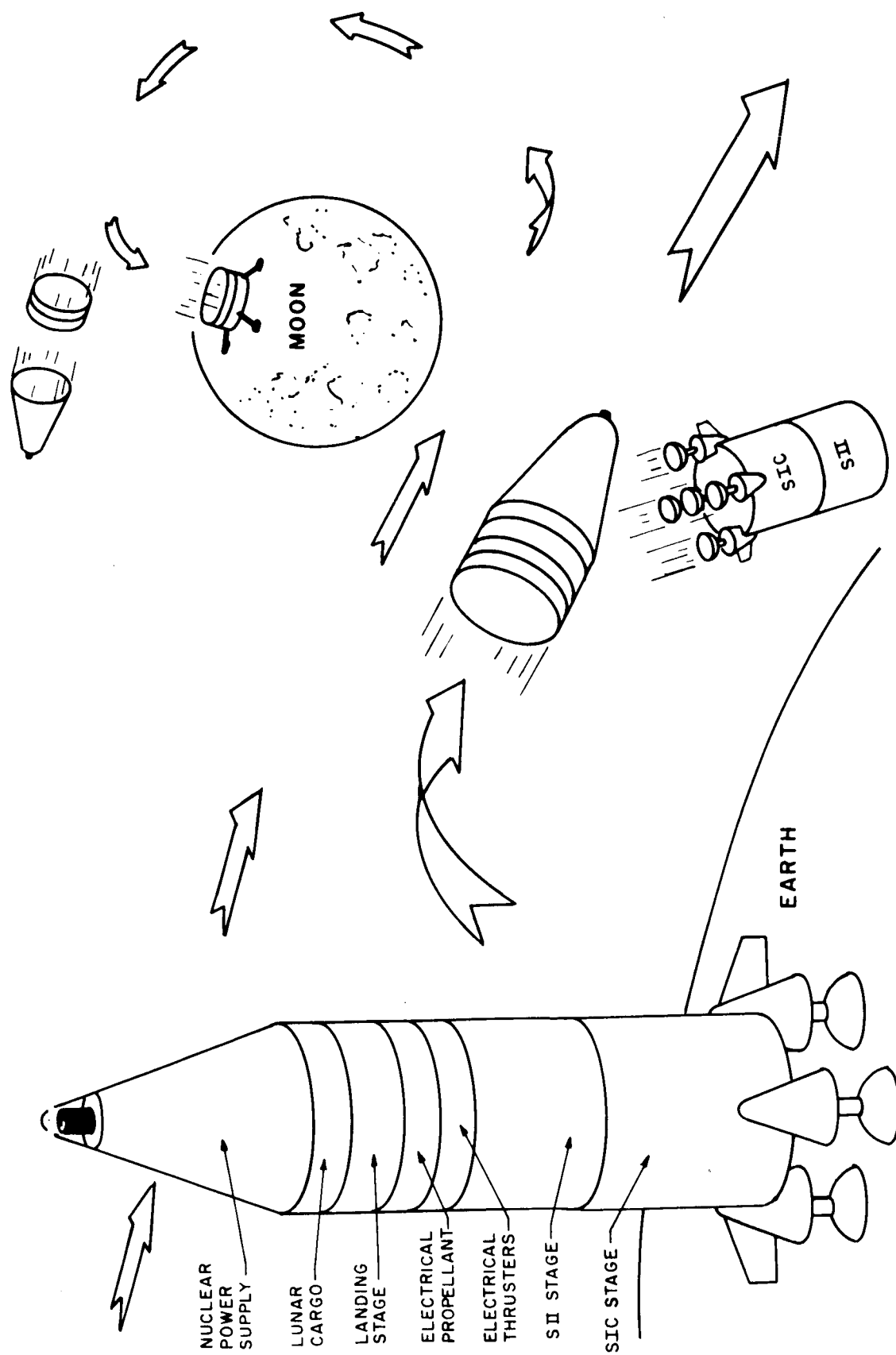


Figure 3-2. Single-Trip Operational Mode with Nuclear Power Source

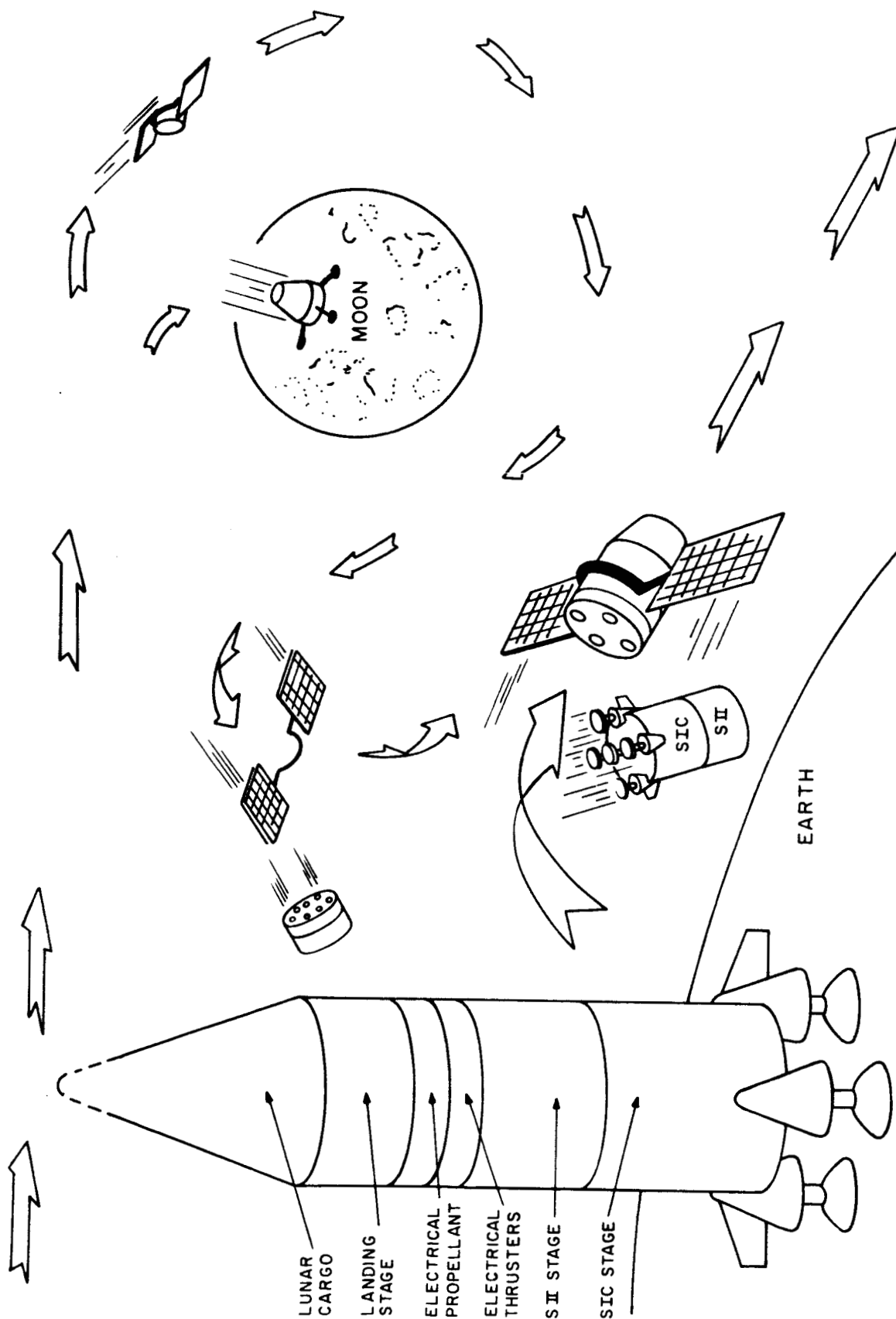


Figure 3-3. Shuttle Operational Mode with Solar Power Source

then proceeds to lunar orbit where the lunar landing vehicle is released. The remaining portion of the electrically-propelled vehicle then returns to earth orbit ready for another trip.

The travel time for the earth to lunar orbit transfer can range between 1000 and 10,000 hours. The life of the electrical power supply can range between 5000 and 50,000 hours. Thus, the placement of the electrical power supply into earth orbit can be significant in the mission performance optimization and needs to be included in the operational mode description.

Two approaches can be used for the shuttle operation. In one, the shuttle operation has a beginning and an end which coincide with the launch and the wear-out of the electrical power supply. The electrical power supply is launched on the first booster along with propellant and lunar landing craft. The electrically propelled vehicle travels to the lunar orbit, releases the lunar landing craft, returns to Earth orbit for rendezvous with a newly launched propellant and lunar landing craft package, and repeats the operation until the electrical power supply is worn out and abandoned. This type of operation is illustrated in Figure 3-4. (The powerplant could also have been launched initially in a separate launch.)

The second type of operation involves the use of multiple electrical power supplies in a cyclic pattern. Two or more powerplants are used to transport the lunar landing craft from earth orbit to lunar orbit. As powerplants fail or wear out, they are abandoned in lunar orbit. The "good" powerplants return to Earth orbit and rendezvous with a package containing a replacement powerplant, as well as propellant and lunar landing craft. This type of operation is shown in Figure 3-5.

In the present study only the single-use Saturn V class of booster is considered for the first transportation step, i.e., ascent from the earth's surface to circular orbit. The key influence of the earth launch system on lunar logistics is the cost per unit mass to orbit. Additional influences are booster payload mass, geometry, center of gravity,



Figure 3-5. Mission Profile for Modularized Nuclear-Electric Powerplant

interface acceleration pattern, ground handling procedures and launch procedure. The results achieved, based on the use of the Saturn V, can generally be converted to application to other boosters of interest.

The operational variations for the orbit transfer phase involve many practical considerations. The one-way trip mode requires abandonment of the electrical power supply in lunar orbit after each mission. However, this scheme is mechanically least complicated because space rendezvous and assembly are not required. The single powerplant shuttle not only requires orbital mating but has an inherent matching difficulty between its initial voyage and subsequent voyages because the powerplant in the initial launch displaces either electric system propellant or lunar landing craft. This situation is improved by launching two vehicles to mate in earth orbit for the first outbound voyage. Because the number of powerplant trips is expected to be limited to about four (as a result of the powerplant life limitation), the initial voyage is quite influential in establishing optimum powerplant and landing vehicle size. The use of multiple powerplants is a second approach towards finding a more favorable powerplant and landing vehicle size compromise. More than two powerplants are probably not of interest in the case of nuclear power sources because of the configuration problems brought about by the need for nuclear radiation shielding.

In both the single and multiple powerplant shuttle operational modes, many options exist for disposition of electrical thrusters and propellant tanks. These options are:

- (1) Maintaining constant landing vehicle size between successive voyages and allowing thruster jet velocity to vary
- (2) Maintaining constant thruster jet velocity between successive voyages and allowing landing vehicle size to vary
- (3) Allow both landing vehicle size and thruster jet velocity to vary between successive voyages to minimize transportation cost at constant powerplant life (or to minimize powerplant life at constant transportation cost)

- (4) Utilization of either constant, or variable specific impulse thrusters permanently mated to the powerplant (requiring in-space fluid line connections)
- (5) Utilization of separate electrical thrusters for the outbound orbit transfer permanently mated to the outbound propellant tanks, which are discarded in lunar orbit and replaced in earth orbit.
- (6) Utilization of inbound electrical thrusters permanently mated to the powerplant requiring in-flight fluid line connections for the inbound thrusters only.
- (7) Utilization of inbound electrical thrusters mated to the inbound propellant tanks abandoned in either earth orbit or later abandoned in lunar orbit, in either case thruster and tank being replaced in earth orbit.
- (8) Utilization of inbound electrical thruster and propellant tank permanently mated to the powerplant and sized to accomplish all of the inbound voyages eliminating re-supply requirements.

The above options for the lunar logistic operational mode yield a large number of space vehicle configurations, which are classified by Figure 3-6. In the present study, performance estimates have been generated for the best of these many combinations.

The third and last transportation phase to be considered is the descent from lunar orbit to the lunar surface. A high thrust propulsion system such as nuclear or chemical rockets is necessary. Only the chemical rocket has been considered to date. The multiple-use lander appears to be at a disadvantage in terms of cargo delivery efficiency when propellants are supplied from the earth for both descent and ascent phases. The value of a multiple-use lander is the savings in not having to repetitively transport landing equipment and structure from the earth. However, reuse requires the transportation of additional propellant for a portion of the descent phase and for the entire ascent phase. This supplied material will exceed the mass of the dry landing vehicle

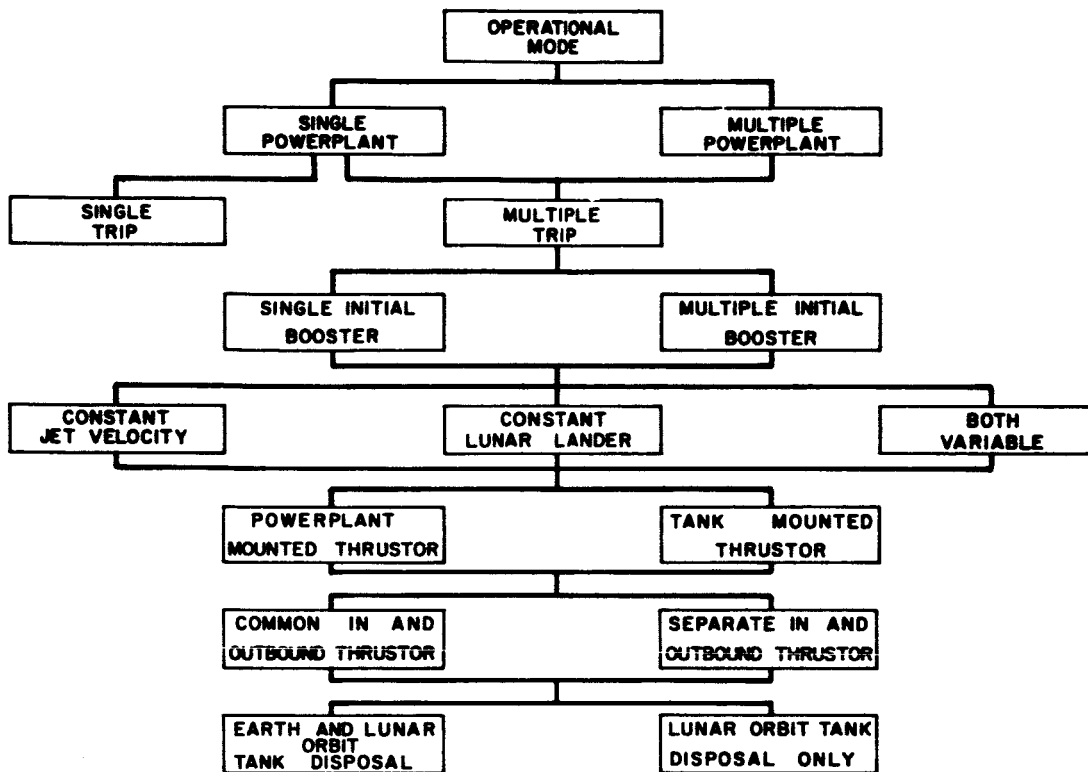


Figure 3-6. Lunar Cargo Vehicle Options

The exploitation of lunar resources to manufacture chemical propellants will alter this conclusion and make the re-useable lunar lander the better approach.

The electrically-propelled cargo vehicle can be used in conjunctive missions with high thrust rocket powered cislunar space transportation vehicles similar to that for Apollo. The slow trip time for the electrically-propelled space vehicle makes its direct use rather unattractive for personnel transportation. However, it can deliver unmanned space vehicles to lunar orbit for later use by space travelers. The delivered space vehicle could be a lunar landing vehicle for transportation of personnel from lunar orbit to surface. This scheme is illustrated in Figure 3-7. The electrically propelled vehicle is used to transport a large lunar landing vehicle from earth orbit to lunar orbit. After this vehicle reaches lunar orbit, a second Saturn V launches a chemical rocket propelled Apollo type system to rendezvous with the electrically propelled vehicle in lunar orbit. The crew transfers to the lunar lander brought over by the electrical

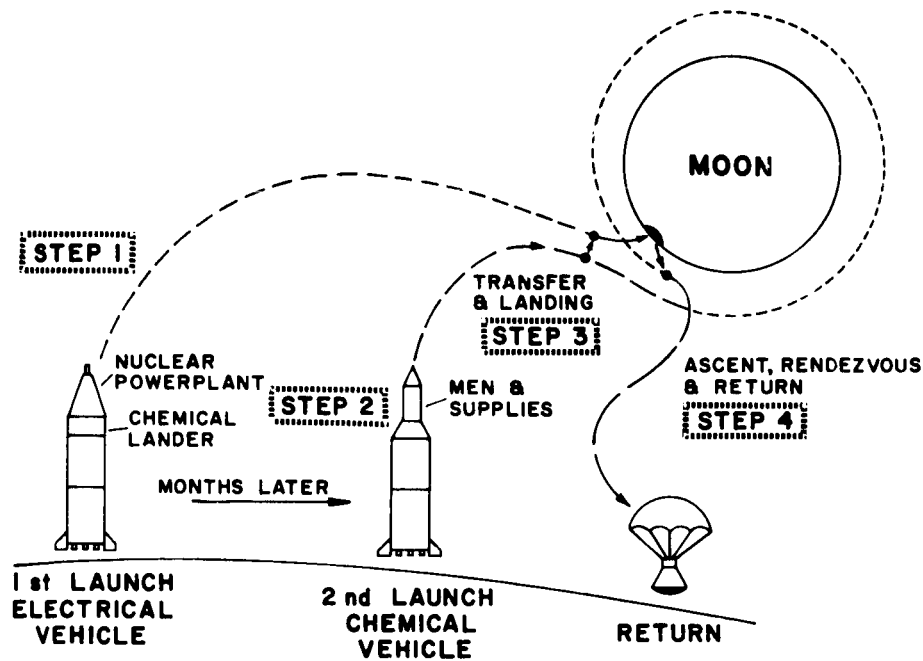


Figure 3-7. Lunar Expedition Strategy

propulsion system and descends to the lunar surface. This approach is attractive because the lunar landing vehicle is equivalent in size to the orbit transfer phase for personnel. The orbit transfer phase cannot be readily done away with. The elimination of the lunar landing stage from the fast trip manned chemical rocket system provides a payload margin for increasing the size of the command and service modules to accommodate more personnel and supplies. The transportation efficiency, i.e., mass per personnel, can be expected to improve with increased size. Thus, the Saturn V personnel transportation capability can increase substantially enabling the crew of a rather sizeable expedition to be transported via a single vehicle.

SECTION IV
PROPULSION REQUIREMENTS

SECTION 4

PROPULSION REQUIREMENTS

The continuous expulsion of a propellant from a space vehicle causes a force of reaction which imparts an acceleration to the space vehicle, proportional to the ratio of thrust to space vehicle mass (Newtonian mechanics). The result of the integrated acceleration over a period of time is a change in the energy level of the space vehicle orbit, which can be expressed in terms of velocity and position. In high thrust propulsion the time period of propulsion is on the order of minutes and the space vehicle position does not change appreciably. Thus, the acceleration force produces a velocity increase alone, and the relationship describing the velocity change is the following:

$$V_C = - V_J \ln (1 - F_P) \quad (1)$$

where

V_C = velocity change of space vehicle, or characteristic velocity

V_J = thruster jet velocity

F_P = propellant fraction = (propellant mass)/(spacecraft initial mass)

Electrical propulsion provides a low thrust for a long period of time, wherein the orbit changes significantly and a portion of the thrust force is applied to potential energy change as well as kinetic energy change. However, the characteristic velocity of the above equation can be utilized for electrical propulsion as a convenient mathematical tool for performing propulsion analysis. The characteristic velocity has been determined to be primarily a function of the initial and terminal orbits, and only slightly varies with acceleration rate (thrust/mass) and thruster jet velocity.

The analysis of the trajectory for the earth-orbit to lunar-orbit passage of an electrically-propelled space vehicle was performed by considering the individual two-body

problems of earth-vehicle and moon-vehicle trajectory characteristics and by patching the two together at an earth-moon transition point. The results of these individual studies were used to develop an empirical model of the overall earth-moon transfer problem as a function of pertinent propulsion system and geometric parameters. The multi-variable LEADER optimization process (Reference 1) was then used to identify the functional variation of the optimum transfer propulsion requirements.

The trajectory pattern for electrical propulsion with a nuclear power source differs from that for a solar power source due to the earth shadow effect. Analyses have been performed for both of these situations. The details of the continuous propulsion trajectory associated with nuclear power systems was presented in Reference 2. The details of the trajectory study for solar power systems have not been previously documented and are presented in the Appendix. The results of the trajectory analyses are presented below.

The characteristic velocity for the earth-orbit to lunar-orbit transfer is approximately 7.8 km/sec for electrical propulsion compared to 4.2 km/sec for high thrust propulsion. This characteristic velocity is based on an initial altitude of 550 kilometers at earth and a final altitude of 37 kilometers at the moon. The characteristic velocity variation with acceleration force is presented in Figure 4-1 for the range of consideration for electrical propulsion. This data applies to the case of continuous electrical propulsion associated with nuclear power systems. The penalty associated with solar power systems due to the earth's shadow has been determined to be a 3 percent increase in characteristic velocity.

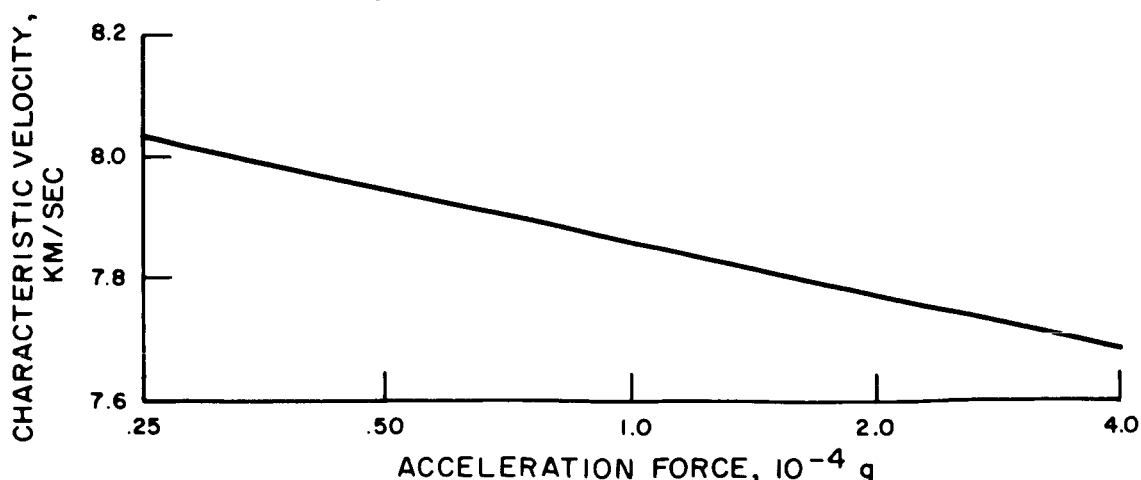


Figure 4-1. Characteristic Velocity Variation with Acceleration Force

The propulsion time required to achieve the required characteristic velocity at a selected thruster jet velocity is determined by the following equation.

$$T = \frac{F_P V_J^2}{7200 g q \eta (P/M)} \quad (2)$$

where

- T = propulsion time, hrs
- F_P = propellant fraction (determined from equation 1)
- V_J = thruster jet velocity, km/sec
- g = gravitational constant = 0.0098 km/sec
- q = conversion factor = 0.102 kg-km/kw
- η = thruster efficiency
- P = thruster power input, KW_e
- M = spacecraft initial mass, kg

The equations (1) and (2) show that the thruster jet velocity has a great effect on both the propellant fraction and the propulsion time. This selection is optimized in the mission analysis portion of the study. The mission model includes additional considerations of powerplant specific weight, travel time constraints, thruster efficiency variation and costs. This allows for systematic selection of powerplant size in addition to thruster jet velocity.

The travel time differs from the propulsion time by the required coast periods. In the case of nuclear power systems, a coast period has been found to be desirable for best overall performance, and this coast period is about 100 hours. The propulsion time is in excess of 1000 hours; thus, the coast period is not significant to the overall voyage. The solar power system differs because of the shadow effect on the solar array by the earth on every orbit revolution. At very low earth orbits the propulsion portion of the orbit is 50 percent, and this portion increases with higher orbits. The integration of this shadow effect over the entire orbit transfer yields an 18 percent increase in trip time for solar power systems compared to nuclear power systems.

SECTION V
SPACECRAFT CHARACTERISTICS

SECTION 5

SPACECRAFT CHARACTERISTICS

The electrically-propelled cargo vehicle is comprised of the component parts as shown in Figure 5-1. The characteristics of the Saturn V were assumed for the earth launch vehicle. The particular Saturn V characteristics of interest and the values employed for the study are the following:

1. Payload injected into a 550-km circular earth orbit — 109,000 kg
2. Payload diameter — 10 meters
3. Payload height — 39 meters

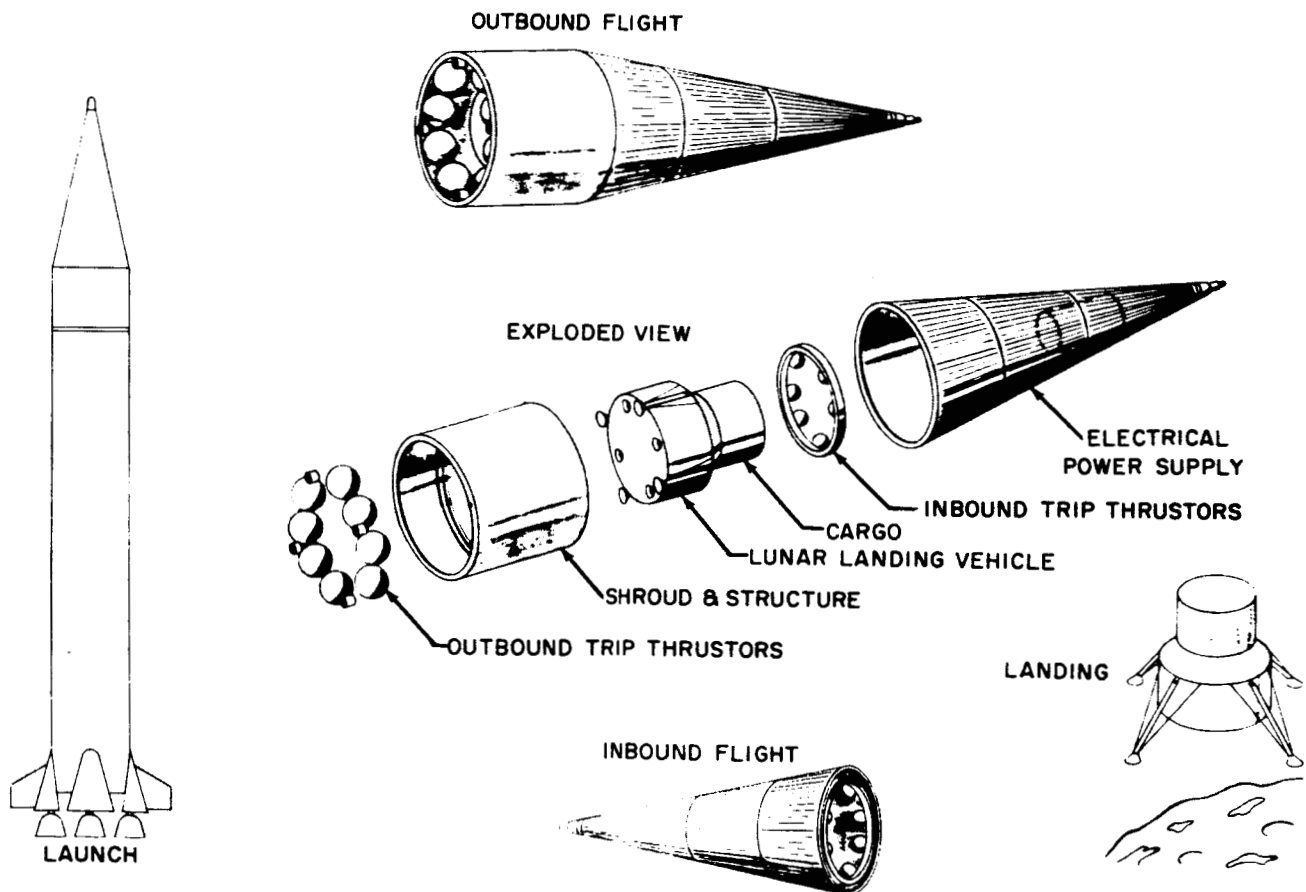


Figure 5-1. Conceptual View of Single Powerplant Lunar Cargo Shuttle

The lunar landing vehicle characteristics were based on the use of hydrogen-oxygen rocket thrustors and cryogenic storage. A preliminary analysis performed (reported in Reference 2) showed that a cargo fraction of 0.44 to 0.46 could be attained, based on a lunar landing vehicle initial mass in earth orbit between 50,000 and 80,000 kilograms.

The nuclear powerplant was studied in the most detail, and the results were reported in Reference 3. The study included Rankine, Brayton and thermionic systems. For each, three levels of state-of-the-art were assumed. The features of these selected power systems are listed in Table 5-1. The state-of-the-art are designated "Early", "Intermediate", and "Advanced". These terms are relative in that a so-called Early system might not be available for first flight until after 1975. These classifications are associated with selected reactor temperatures and conversion efficiencies. For each system parametric performance characteristics have been generated to be used for power supply comparison and mission evaluation.

TABLE 5-1. CHARACTERISTICS OF ASSUMED NUCLEAR POWERPLANT CONCEPTS

		RANKINE CYCLE SYSTEM	BRAYTON CYCLE SYSTEM	THERMIONIC SYSTEM
		3 NAK (EARLY) LITHIUM POTASSIUM NAK	1 HELIUM-XENON HELIUM-XENON HELIUM-XENON	2 NAK - NAK
EARLY	NUMBER OF LOOPS			
	REACTOR COOLANT			
	TURBINE WORKING FLUID			
	RADIATOR COOLANT			
INTERMEDIATE	REACTOR FUEL	UC	UO ₂ -W CERMET	-
	TURBINE INLET TEMP (F)	1600	2100	-
	TURBINE EFFICIENCY	0.65	0.87	-
	RADIATOR INLET TEMP (F)	1246	1400 (MAX)	-
	RADIATOR MATERIAL	SS-Cu	Al-SS	-
ADVANCED	REACTOR FUEL	UC	UO ₂ -W CERMET	UO ₂
	TURBINE INLET TEMP (F)	1850	2700	0.1377-10*
	TURBINE EFFICIENCY	0.80	0.89	1190
	RADIATOR INLET TEMP (F)	1246	1400 (MAX)	Be-SS
	RADIATOR MATERIAL	Be-SS	Be-SS	

* CONVERTER EFFICIENCIES DIODE POWER DENSITY, W CM²

The estimated powerplant mass versus net electrical power output determined in this study are presented in Figures 5-2, 5-3, and 5-4 for the Rankine, Brayton and thermionic cycles, respectively. A comparison of the net electrical power provided in a total powerplant package of 11,300 kilogram (25,000 lb) mass is presented in Figure 5-5. These mass estimates include the shield.

The system mass for each power systems was determined by first analyzing the characteristics of each major component of the system. The effects of significant parameters were determined, and where not optimized, suitable values were selected. Since the radiators are the major mass items, they received the most attention. System masses were then assembled at several power levels to determine the variation with power. Power system masses were optimized to minimize total mass without regard to their effect on other systems of the lunar cargo vehicle, i. e. , the lunar lander section, electrical propulsion system, and power conditioning and controls.

The Rankine cycle systems are shown to be lowest in mass for comparable technology levels up to a power level of 2 megawatts. Above 2 megawatts, the thermionic systems appear to be slightly lighter. Although the Rankine cycle systems have a higher efficiency than the thermionic systems, the mass of the turbomachinery and associated power conversion equipment becomes dominant and offsets this efficiency advantage at high power levels. The thermionic converters, on the other hand, are included in the reactor mass and are insignificant in mass.

The comparison presented in Figure 5-5, where powerplant mass is constant at 11,325 kg (25,000 lb), is the most appropriate form for interpreting the mass differences on mission performance. The required propulsion time is approximately inversely proportional to the net electrical power. Thus, the advanced thermionic system at 3 megawatts will perform the earth-orbit to lunar-orbit transfer in 1/3 the time of the 1 megawatt Brayton cycle system.

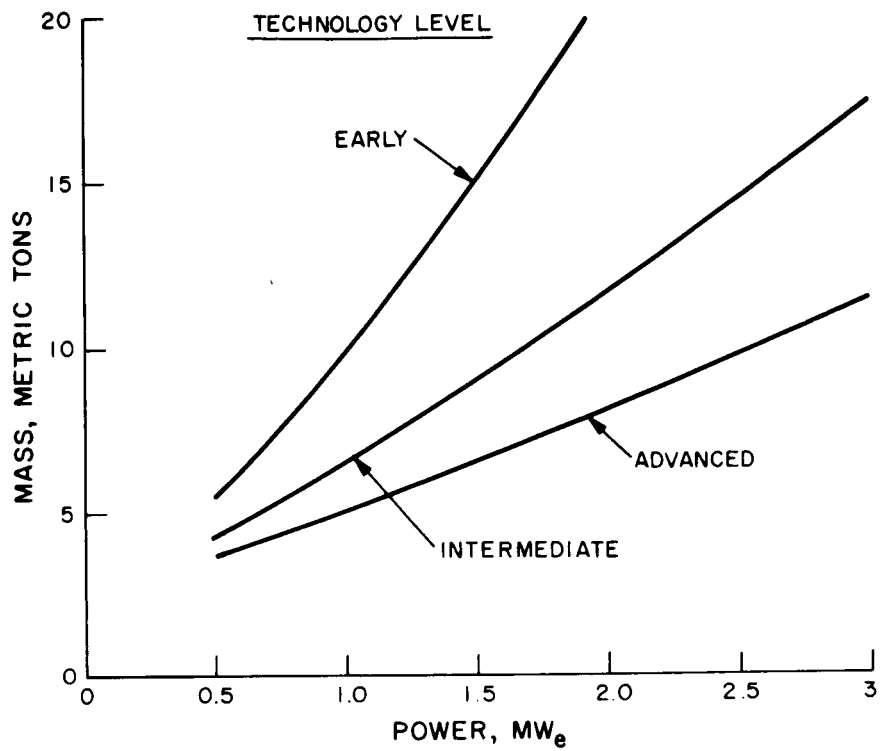


Figure 5-2. Mass vs Power for Rankine Cycle Powerplant Concepts

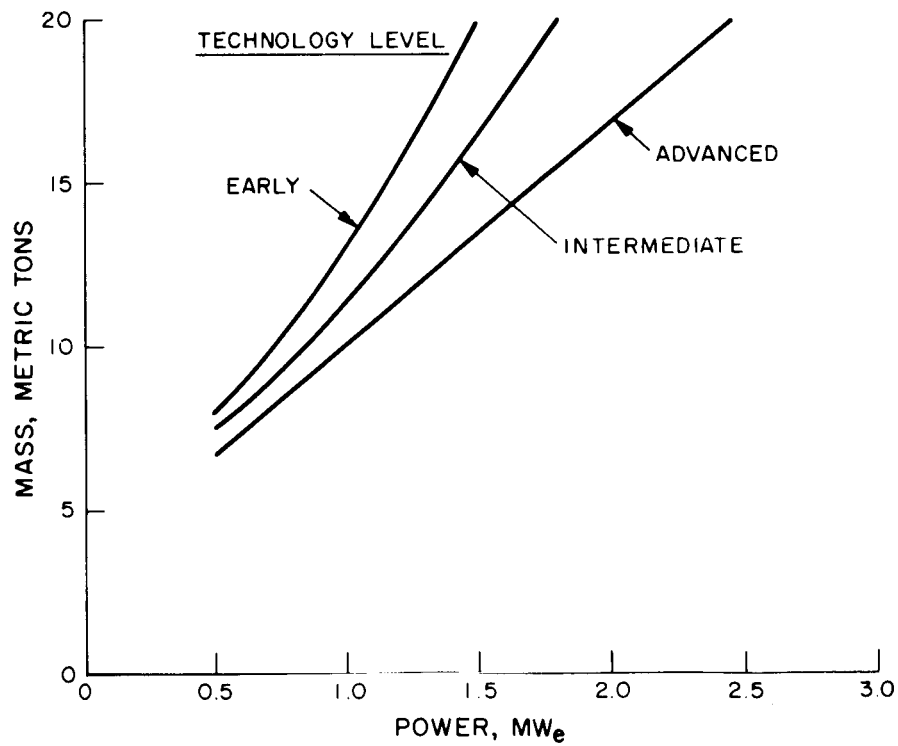


Figure 5-3. Mass vs Power for Brayton Cycle Powerplant Concepts

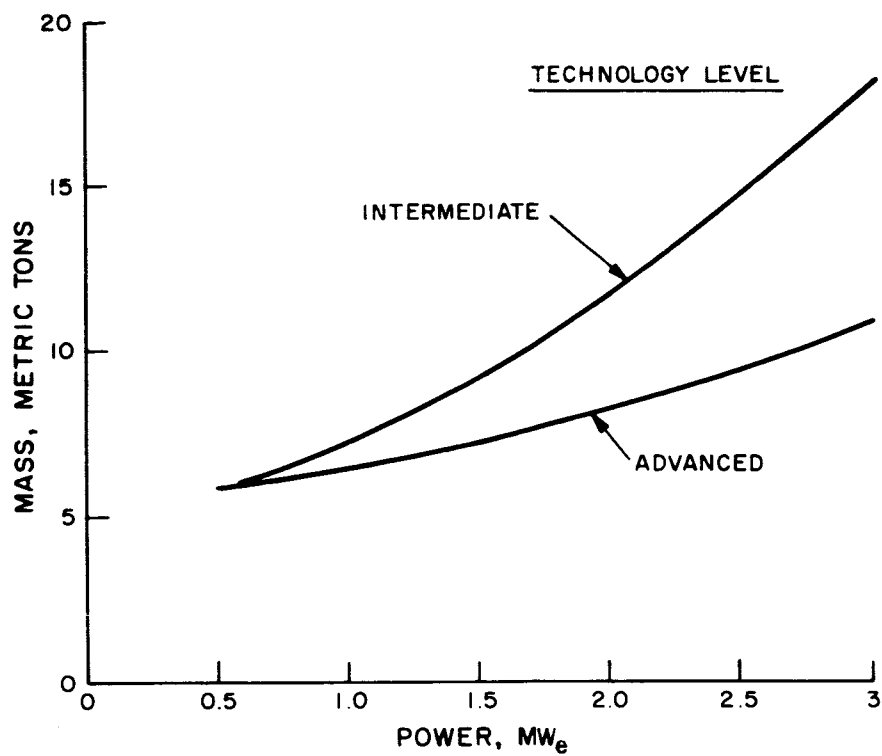


Figure 5-4. Mass vs Power for Thermionic Powerplant Concepts

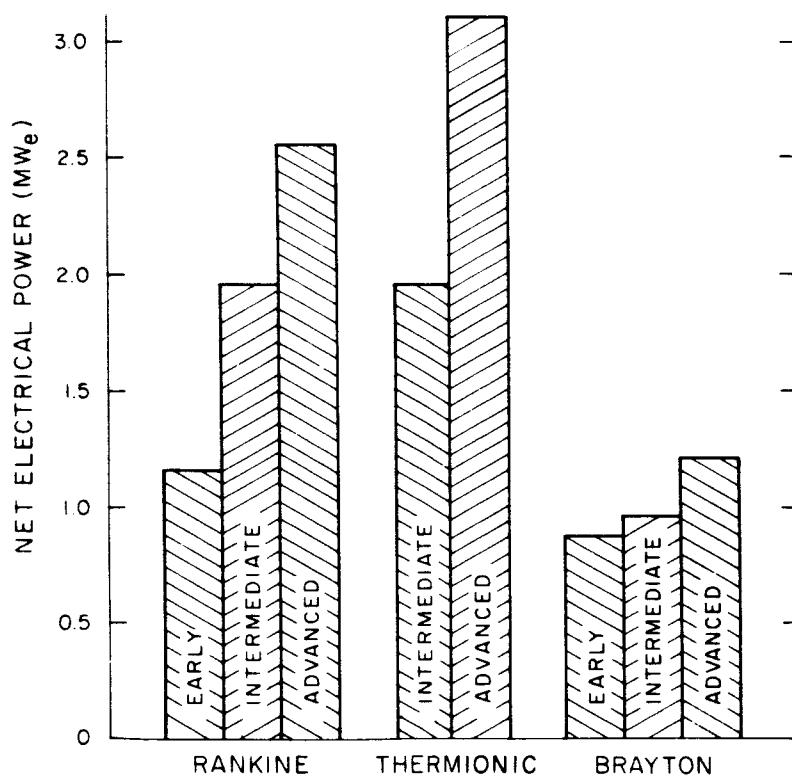


Figure 5-5. Electrical Power Output Comparison at 11,300 Kilogram Total Mass

Based on the results presented in Reference 2, all of the powerplant systems considered here can perform a satisfactory lunar cargo mission. The powerplant specific mass assumed for much of the mission analysis in Reference 2 was 10 KG/KW_e . At this specific mass, electrical propulsion has a substantial advantage over high thrust chemical and nuclear rocket systems. The improved performance capability of the intermediate and advanced Rankine and thermionic system at 4 to 6 KG/KW_e provide extra competitive margin.

The general arrangement of the nuclear powerplant is shown in Figure 5-6 and 5-7 for the Rankine and Brayton cycle powerplants, respectively. The reactor is located near the apex of a conical shape radiator. A shadow shield is provided to protect the powerplant supporting equipment and payload from the nuclear radiation. Figure 5-6 does not show the primary radiator, which is conical in shape and extends as a continuation of the NaK secondary radiator. The thermionic powerplant has a similar arrangement for the location of reactor, shield and radiator. (For more details, see Reference 3.)

The use of solar power for electrical propulsion is of interest because of its earlier availability relative to nuclear. The performance rating of solar power sources has been determined for certain advancements in the state-of-the-art. The solar energy is about 1.3 KW/meter^2 , which can be used to produce $0.1 \text{ KW}_e/\text{meter}^2$ of electrical power by direct conversion in silicon solar cells. Silicon solar cell panels in Mariner II are rated at 90 KG/KW_e (200 LB/KW_e), or 90 grams/meter^2 (2 lb/ft^2); in Mariner IV, at 45 KG/KW_e (100 LB/KW_e), or 45 grams/meter^2 (1 lb/ft^2). In experimental panels 22 KG/KW (50 LB/FT^2), or 22 grams/meter^2 (0.5 lb/ft^2), has been achieved. There is a possibility that the mass can be cut in half again, yielding 11 KG/KW (25 LB/KW) at a panel mass of 11 grams/meter^2 (0.25 lb/ft^2). However, reduced mass can lead to reduced life and this tradeoff needs to be made in a system study.

The solar powered system needs to be sun oriented, whereas the electric thrusters need to be directed generally along the direction of motion. The mechanical design

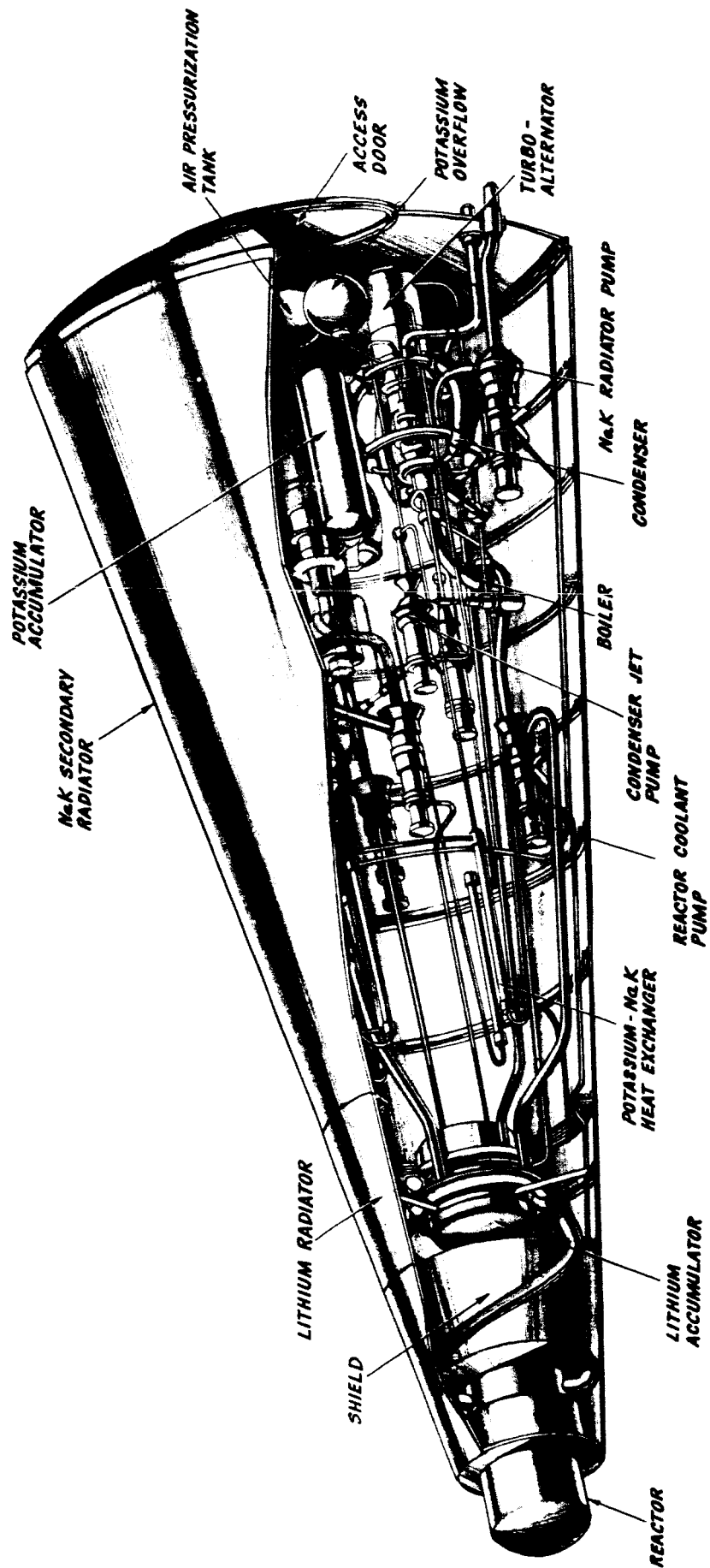


Figure 5-6. General Arrangement of Power Conversion System for Rankine Cycle Nuclear Powerplant

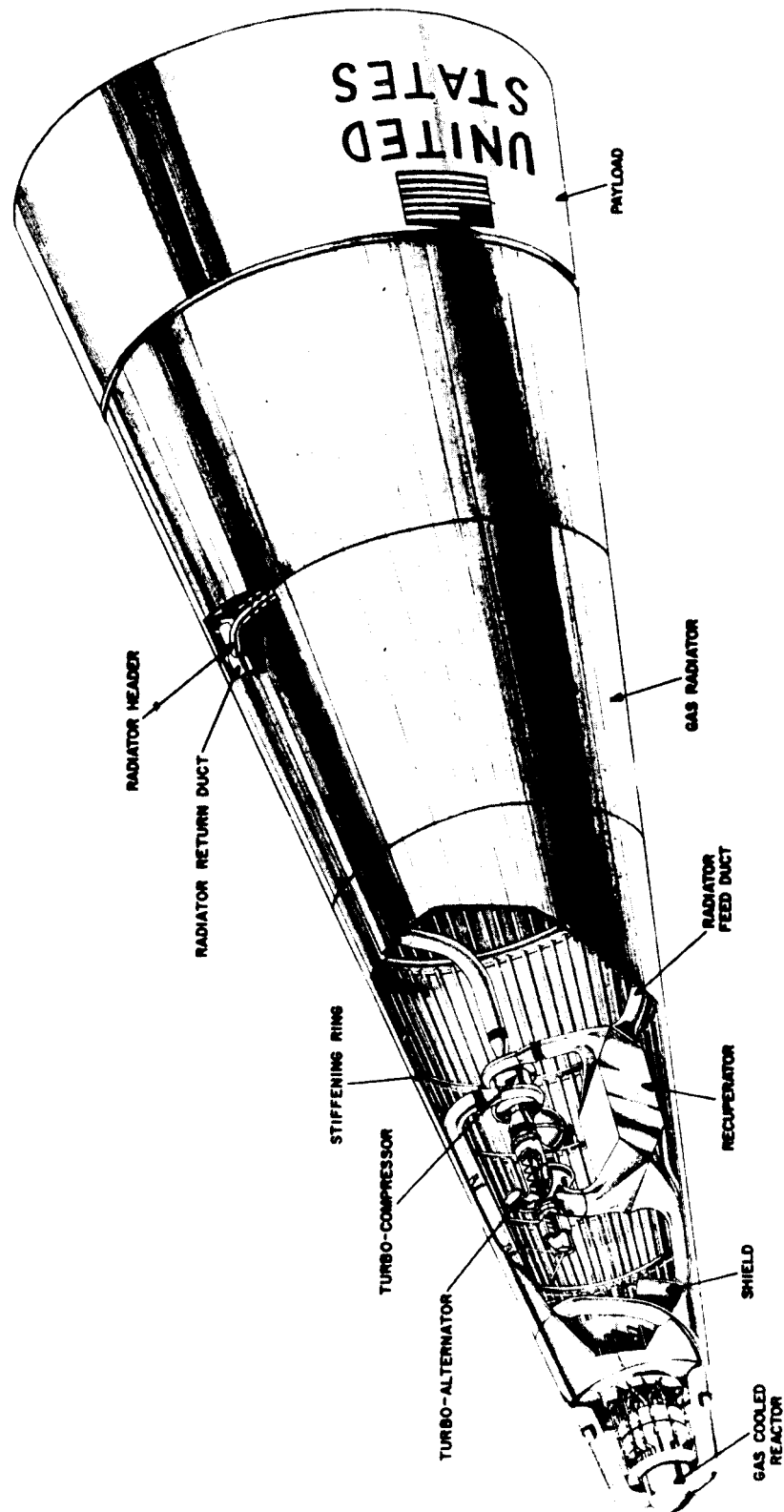


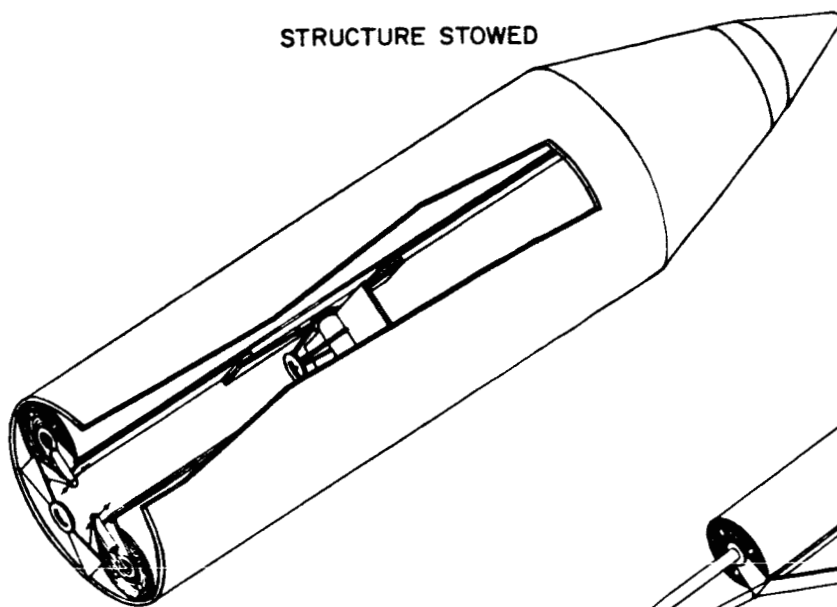
Figure 5-7. General Arrangement of Power Conversion System for Brayton Cycle Nuclear Powerplant

of the spacecraft has not been studied in detail during this study, and needs to be examined to resolve means for accomplishing this requirement and for determining any limits on orientation that would translate into propulsion penalties. The low power density of the solar array leads to a requirement for folding concepts. A one megawatt power system, for example, requires 10,000 square meters of solar panel. Environmental problems also arise. The silicon solar cell is susceptible to radiation damage, and protection against this damage is provided by use of a cover glass. The travel time through the Van Allen belt will set requirements for radiation protection.

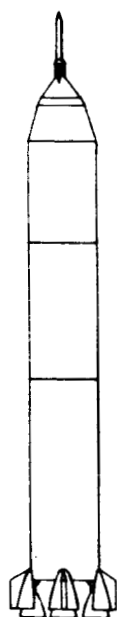
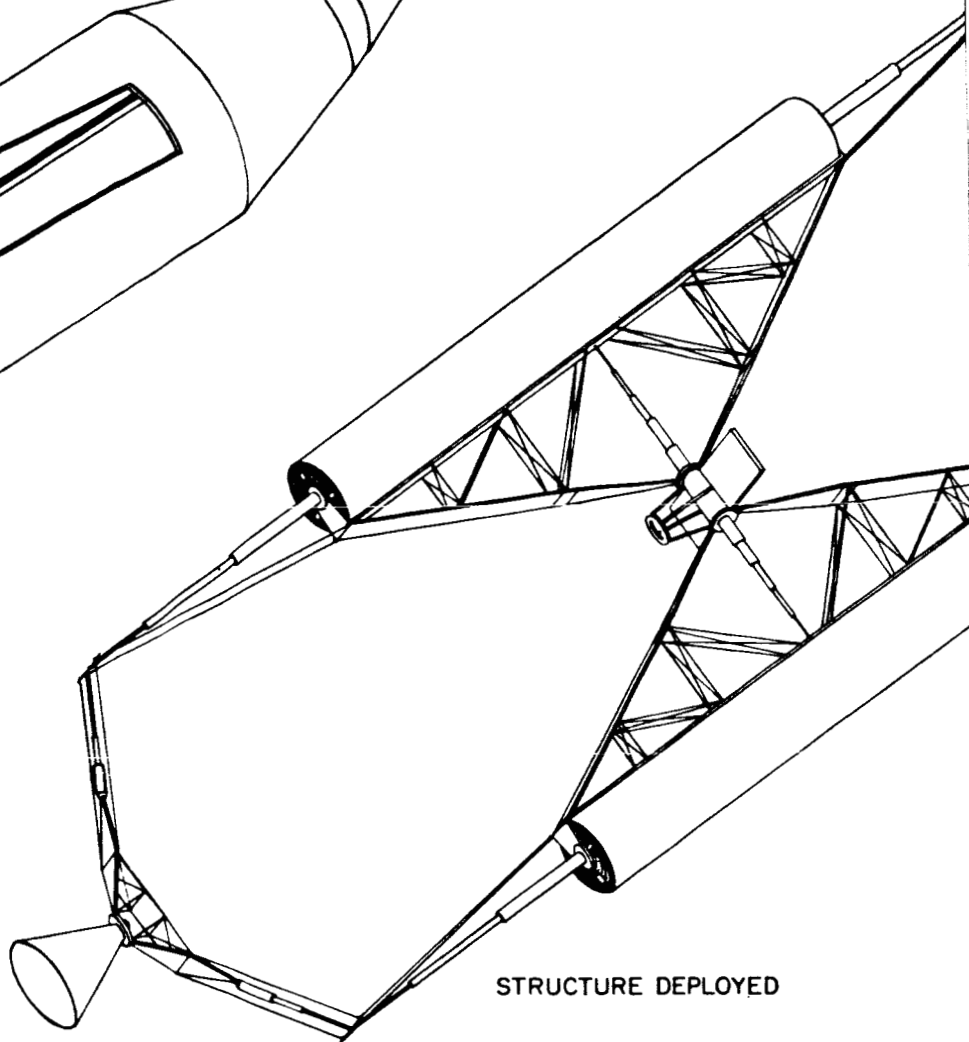
A conceptual design of a solar power system is presented in Figure 5-8. The solar powerplant is launched on a Saturn V vehicle accompanied by a space assembly crew. After launch the shroud is jettisoned, and the primary structure is deployed. The assembly crew then unrolls the solar array to its full length and unfolds it to its full width. The complete spacecraft assembly measures 200 meters (660 ft) by 51 meters (167 ft) for 1.2 megawatts electrical power output. The array surface area varies proportional to power. After checkout of the solar powerplant, the assembly crew returns to earth.

The lunar landing vehicle, propellant and electric thrusters are launched on the next Saturn V booster. This launch could also include a space assembly crew if necessary for the earth orbit rendezvous operation. After orbit is achieved, the lunar landing vehicle is joined to the solar powerplant as shown. In this conceptual approach, the electric thrusters are contained in the lunar landing vehicle. Then, the lunar landing vehicle is oriented to provide the proper thrust vector. During operation the solar powerplant and lunar landing vehicle rotate relative to one another to maintain the proper sun and thrust directions.

STRUCTURE STOWED

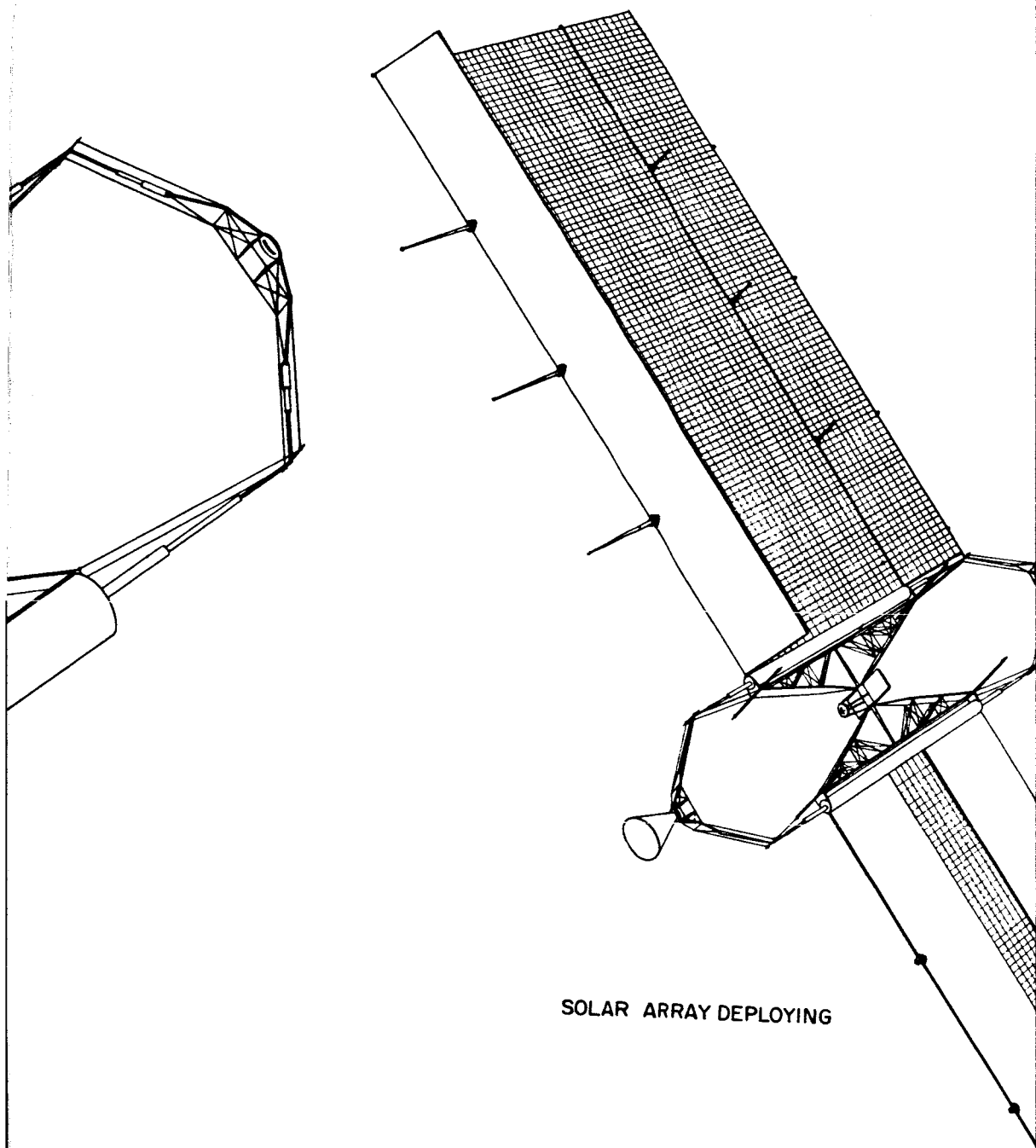


STRUCTURE DEPLOYED



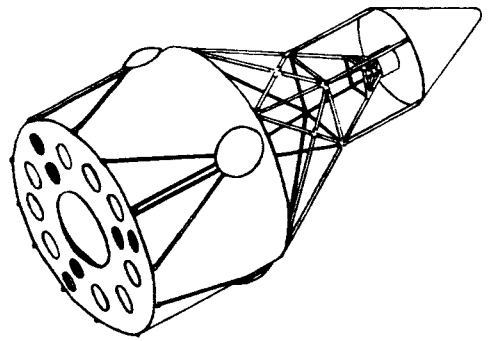
SATURN V

5-11-1

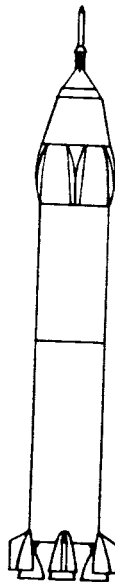


SOLAR ARRAY DEPLOYING

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LUNAR VEHICLE



SATURN V

5-11-3

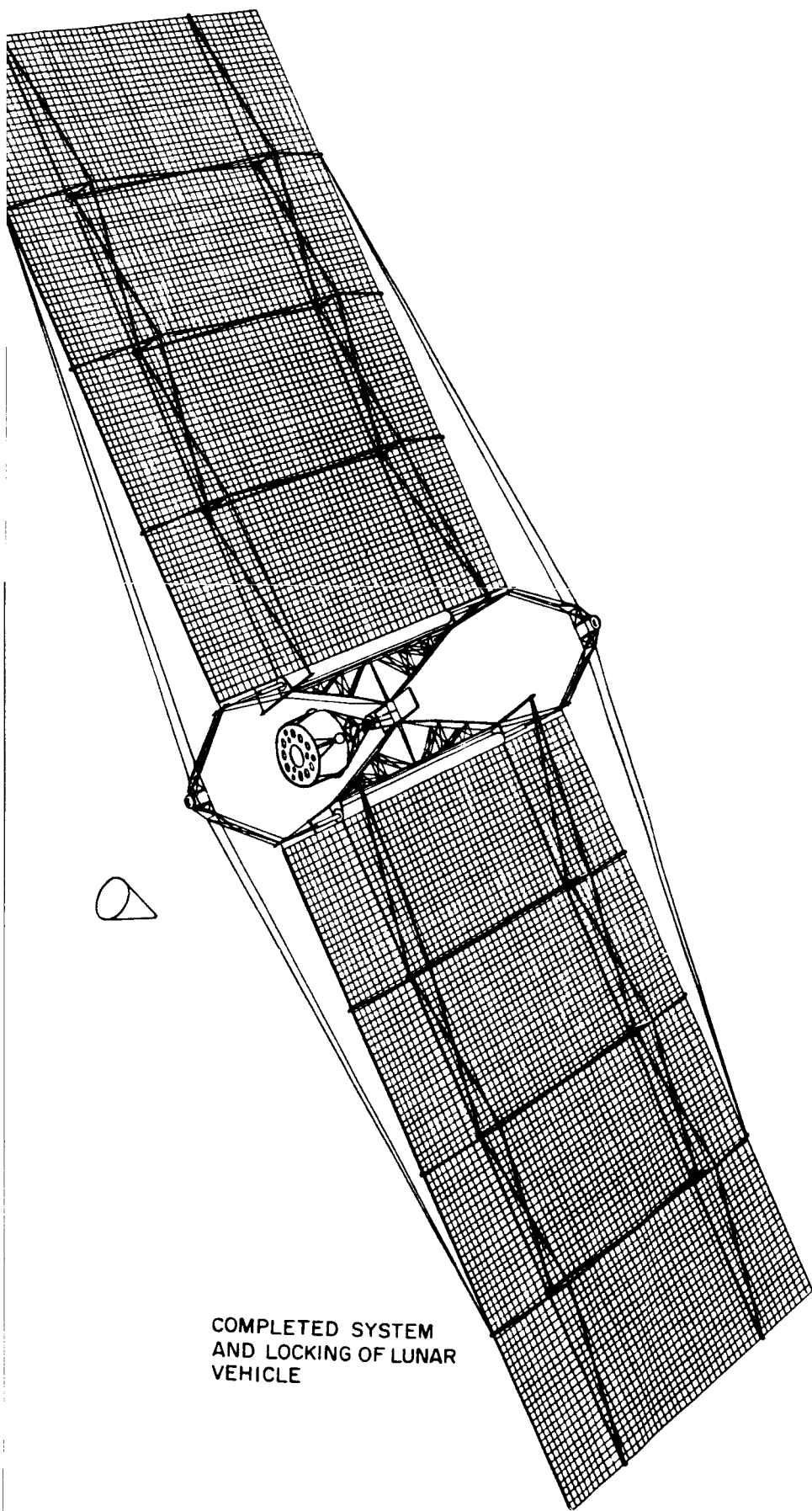


Figure 5-8. Conceptual Design of Solar
Powered, Electrically Pro-
pelled Lunar Cargo Vehicle

The electric thruster type used in the analysis is the electron bombardment ion engine, which was selected because it typifies the ion jet thruster and has received most widespread attention. The efficiency and mass characteristics assumed are presented in Figures 5-9 to 5-11. The mass includes the portion of the power conditioning equipment located with the thruster. The mass of power conditioning equipment located near the nuclear power supply is assumed to be included in the nuclear power supply mass. The propellant tank, structure and insulation is assumed constant at ten percent of the gross propellant mass.

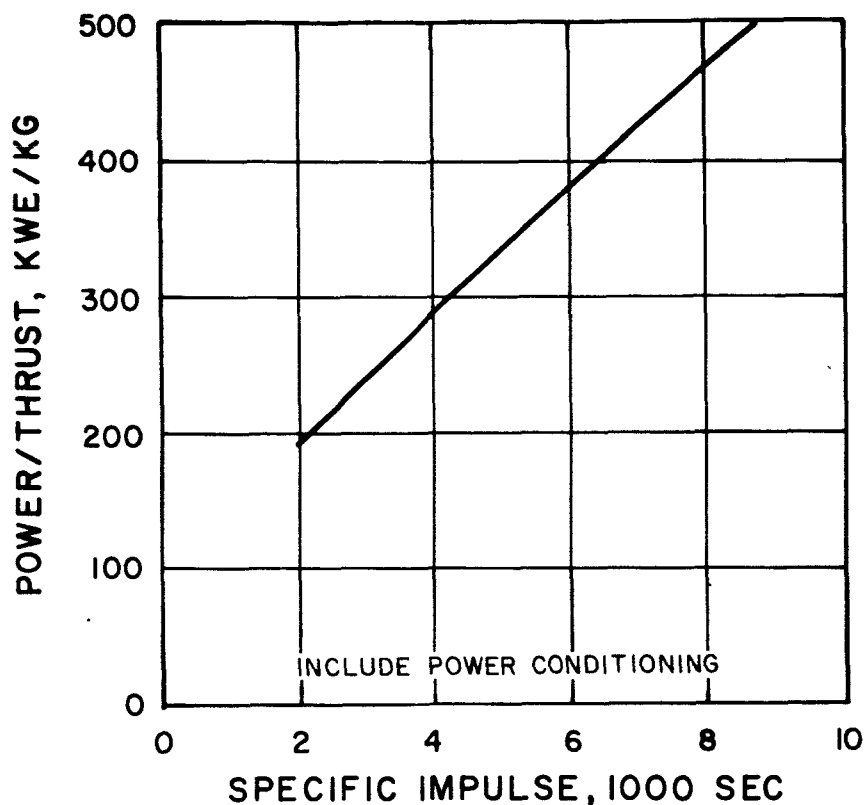


Figure 5-9. Assumed Thruster Power to Thrust Ratio

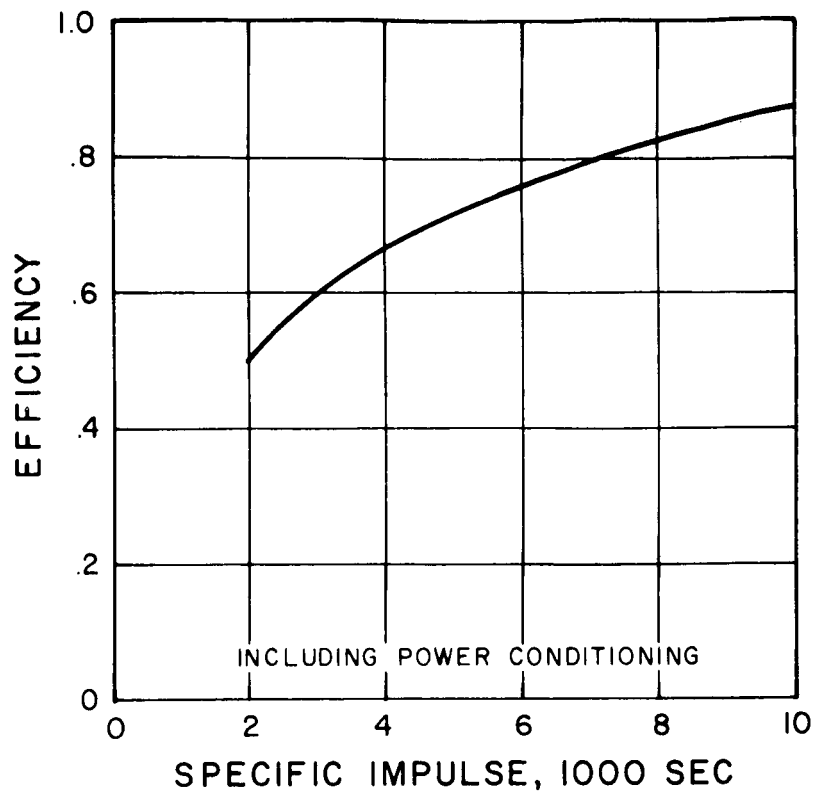


Figure 5-10. Assumed Thruster Efficiency

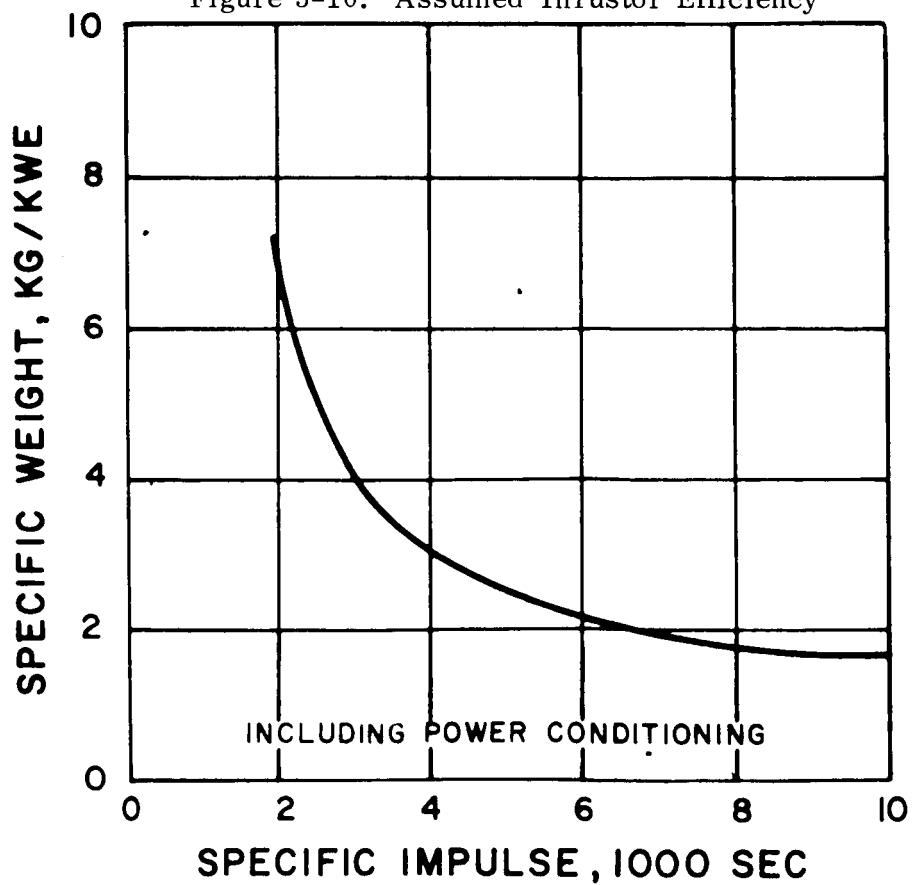


Figure 5-11. Assumed Thruster Specific Weight

SECTION VI
MISSION ANALYSIS

SECTION 6

MISSION ANALYSIS

The mission performance analysis has been repeated several times during the course of the program to keep in pace with the results of concurrent evaluation of the many aspects of the spacecraft and the mission. The initial approach was to develop a generalized model for analysis of a large combination of operating modes such as single-trip, multiple trip and multiple powerplant. Each of these modes had various options regarding launch situation, dry propellant tank disposal and thruster replacement. As a result of the early studies, the number of operational modes of interest was reduced and the study was continued in more detail. The generalized mission analysis was completed during the first year, and results were published in Reference 2. The next step in the program was a detailed study of nuclear powerplant characteristics, which was directed at eight assumed concepts. These results, which were reported in Reference 3, have been formulated into mathematical models for mission evaluation. The solar power source was also introduced into the study, a mission analysis was performed, and a comparison was made with the nuclear systems. The more significant of these mission evaluations are discussed below.

6.1 GENERALIZED ANALYSIS

The major task during the first year of the program was to develop a generalized analysis of the sustained lunar supply system based on the use of a reusable, power-limited vehicle or propulsion module. Results of this analysis were developed in parametric form and reported in Reference 2. The major performance characteristics were determined for various state-of-the-art assumptions. A summary of these results and a discussion of the background considerations are presented below.

The electrically-propelled cargo vehicle was analyzed in relationship to the chemical rocket performance capability. The selection criteria for component optimization was minimum cost, and a "Cost Index" was used as a parameter of merit, which is defined as the ratio of nuclear-electric system to chemical system total costs to transport a selected quantity of cargo from the earth to the lunar surface. The criteria of using minimum cost rather than maximum cargo delivered was found to be a key factor in performing the analysis, and this led to consideration of substantially smaller powerplants than was used in previous studies.

The particular assumptions used in the cost model are presented below. However, it is the ratio of costs that is important for purpose of system optimization.

- (1) Booster Cost (C_B) - \$60,000,000 per Saturn V launch to place a payload of 109,000 kg in a 550-km circular orbit about the Earth. This results in a booster cost of \$550/kg of payload.
- (2) Powerplant Cost (C_Q) - \$1100/kg for the nuclear-electric powerplant, electric thrusters, and associated power conditioning systems. Thus, the ratio of nominal powerplant to booster costs is 2:1.
- (3) Propellant Costs (C_P) - \$44/kg for the electric thruster propellant requirements.

The basic cost index can, therefore, be obtained from the equation:

$$C_I = \frac{(C_B + w P C_Q + W_P C_P)/W_N}{C_B/W_C}$$

where W_C = the lunar payload capability with chemical rocket propulsion, kg

W_N = the lunar payload capability with electrical propulsion, kg

P = power output of nuclear power supply, KW_e

W_P = propellant mass, kg

w = specific weight of nuclear power supply, KG/KW_e

Parametric studies indicate the effects of variations in the above powerplant to booster cost ratio from 1 to 4. These variations, therefore, include the range of Saturn V costs from \$30,000,000 to \$120,000,000, the range of powerplant costs from \$500 to \$2000 per kilogram, or combinations of the two. The reference Saturn V payload delivery capability is 12,700 kilogram. No additional costs have been included, however, for the upper stage propulsion beyond Earth orbit for the transfer and soft landing operations. Consequently, a cost index of 1.0 implies an actual payload cost of \$4600 per kilogram. The approach described above only accounts for the manufacturing cost of nuclear power supply, which is adequate for the first order system optimization.

A more refined analysis has been derived, which includes development cost, and this is described in Table 6-1. This cost model has been applied in the more recent mission studies, and it also contains many rather simplified assumptions. The number of development power plants is related to the demonstrated reliability. The total development cost is proportional to the product of power rating, life rating and number of development power plant assemblies. A learning curve is applied to the procurement cost. The total lunar cargo requirement anticipated is used for the determination of development cost depreciation. The consideration of powerplant failures and consequent payload losses is included in the cost index. The use of this method allows a rather broad and generalized cost analysis to be performed for the electrically-propelled cargo vehicle.

A method to penalize the performance of the electrically propelled vehicle for long trip times has been devised, which is based on the probability of vehicle losses en-route to the moon. (Total trip time as used herein is synonymous with life rating of electrical power supply and propulsion system.) The vehicle loss penalty is a cost penalty added to the basic cost index to reflect the probability of a powerplant failure and attendant loss of payload during an out-bound Earth-Moon transfer or the probability of an inbound powerplant failure which would require replacement before

TABLE 6-1. COST MODEL FOR NUCLEAR SYSTEM

EQUATIONS	NOMENCLATURE
$Z_{NQD} = \frac{R_{EL}}{1 - R_{EL}}$ $Z_{CQD} = \frac{Z_{NQD} (1 + \ell_n R_L / \ell_n^2) + \ell_n R_L / \ell_n^2}{1 + \ell_n R_L / \ell_n^2}$ $X_C = 1 - \left(1 - e^{-T_O / T_{MF}}\right)^{X_F}$ $Z_{NBT} = \frac{W_C}{W_{CN} X_C R_{EL}}$ $Z_{NQT} = Z_{NBT} + Z_{NQD}$ $Z_{CQT} = \frac{Z_{NQT} (1 + \ell_n R_L / \ell_n^2) + \ell_n R_L / \ell_n^2}{1 + R_L / \ell_n^2}$ $C_T = C_R + X_{ND} S_{CQ} P Z_{CQD} (T_O / 10,000) + S_{CQ} P Z_{CQT} + Z_{NBT} C_B$ $C_I = \frac{W_{CC} C_T}{C_B W_C}$	C_B - BOOSTER COST, \$ MILLION C_I - COST INDEX, RELATIVE TO CHEMICAL C_R - COMPONENT R & D COST, \$ MILLION C_T - CUMULATIVE TOTAL MISSION COST, \$ MILLION P - NET POWER TO THRUSTERS, MW _e R_{EL} - POWER SOURCE QUALIFIED RELIABILITY R_L - POWER SOURCE COST LEARNING RATE S_{CQ} - FIRST POWER SOURCE COST, \$ /WATT T_{MF} - MEAN TIME TO RADIATOR FAILURE, HR T_O - TRIP TIME, HR W_C - CUMULATIVE LUNAR CARGO, TONS W_{CC} - REFERENCE CHEMICAL SYSTEM CARGO, TONS W_{CN} - NUCLEAR SYSTEM CARGO, TONS X_C - RADIATOR NO-FAILURE PROBABILITY X_F - RADIATOR FAILURES TO ABORT X_{ND} - DEVELOPMENT/MANUFACTURE POWER SOURCE COST Z_{CQD} - CUMULATIVE/INITIAL DEV POWER SOURCE COST Z_{CQT} - CUMULATIVE/INITIAL TOTAL POWER SOURCE COST Z_{NBT} - CUMULATIVE NO OF BOOSTERS Z_{NQD} - NO OF DEVELOPMENT POWER SOURCES Z_{NQT} - CUMULATIVE NO OF POWER SOURCES

the next outbound leg. The basic assumption used in the development of the survival penalty is the validity of the exponential failure model:

$$R = \exp(-t/\theta) \quad (1)$$

where R is the probability of experiencing a single failure after t hours of operation with a powerplant designed for a mean time to failure of θ hours. The consequences of the failure are dependent upon the degree of redundancy built into the powerplant (number of failures leading to loss of powerplant), the number of engine modules used, and upon whether the failure has occurred during either an outbound or inbound trajectory. A distinction is, therefore, made by this approach between redundancy in which a powerplant is designed to sustain operation until a prescribed number of failures have occurred and modularization in which a number of powerplants are used simultaneously with each individual powerplant lost after a single failure. The survival model has been based upon the assumption that payload delivery can be completed as long as a single operating powerplant module remains.

Similarly, it has been assumed that the powerplant can be returned to the earth for re-use if one operating powerplant module remains. The returned powerplants are re-used for subsequent trips, however, only if they have experienced no previous failures, regardless of the number of failures permitted before the loss of powerplant. This approach results in a survival penalty which is a combination of an out-bound penalty and a corresponding inbound penalty. The implementation of these basic ground rules to each of the various modes of lunar ferry operation were discussed in Reference 2.

The mission analysis procedure employed for the electrically-propelled cargo vehicle is described below in connection with the single-trip operating mode. This procedure is applied in a similar manner to the shuttle operation.

The selection of values for two independent parameters such as vehicle cargo mass and thruster specific impulse yields a closed form solution, assuming that relationships are defined for powerplant mass versus power, thruster mass versus specific impulse, thruster efficiency versus specific impulse, propellant tank masses versus size, and other supporting subsystems versus size. This closed form solution consists of a powerplant size and a required propulsion time. A typical parametric representation of results is presented in Figure 6-1. From this analysis a specific impulse can be selected which minimizes propulsion time as a function of cargo mass. This relationship determines the locus of optimum solutions, if the cost of the powerplant can be ignored. In the case of both nuclear and solar powerplants, the powerplant cost has been found to be quite significant, and it varies almost proportionally with size. Thus, the consideration of powerplant cost will bias the optimum solution towards small powerplant sizes.

This optimization of the electrically-propelled cargo vehicle parameters on the basis of cost is shown in Figure 6-2. A "Cost Index" is estimated as previously defined. Isolines of constant cargo mass are determined by variation of thruster specific impulse. A line is drawn tangent to the cargo mass curves as shown, which

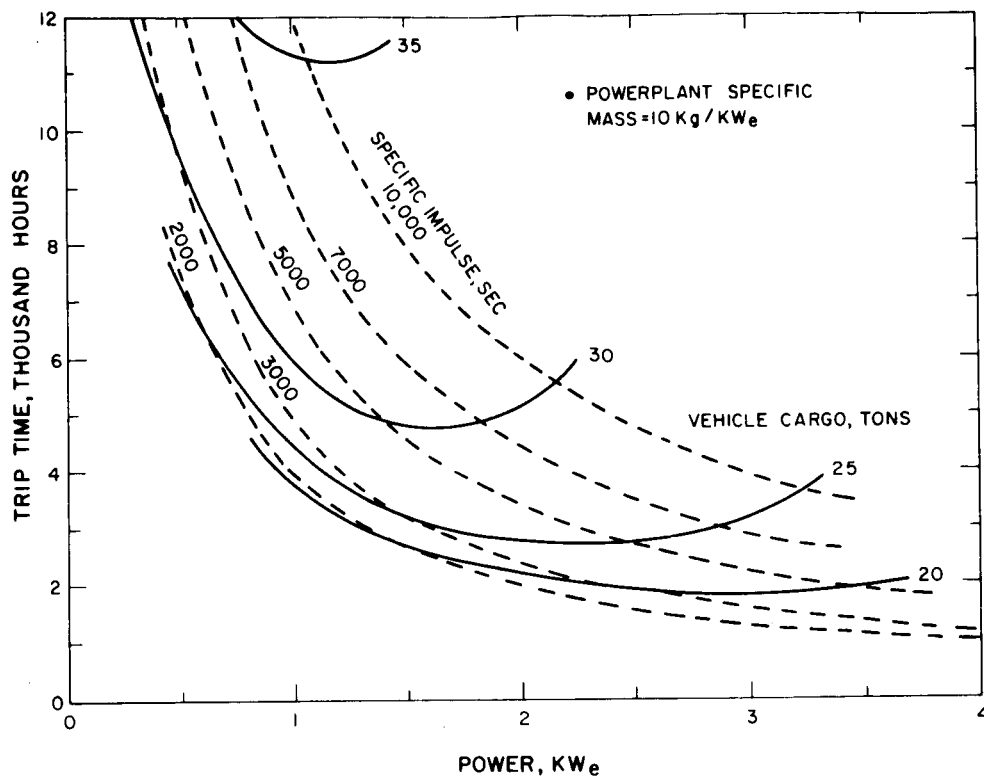


Figure 6-1. General Performance Characteristics of Nuclear System for Single-Trip Operation

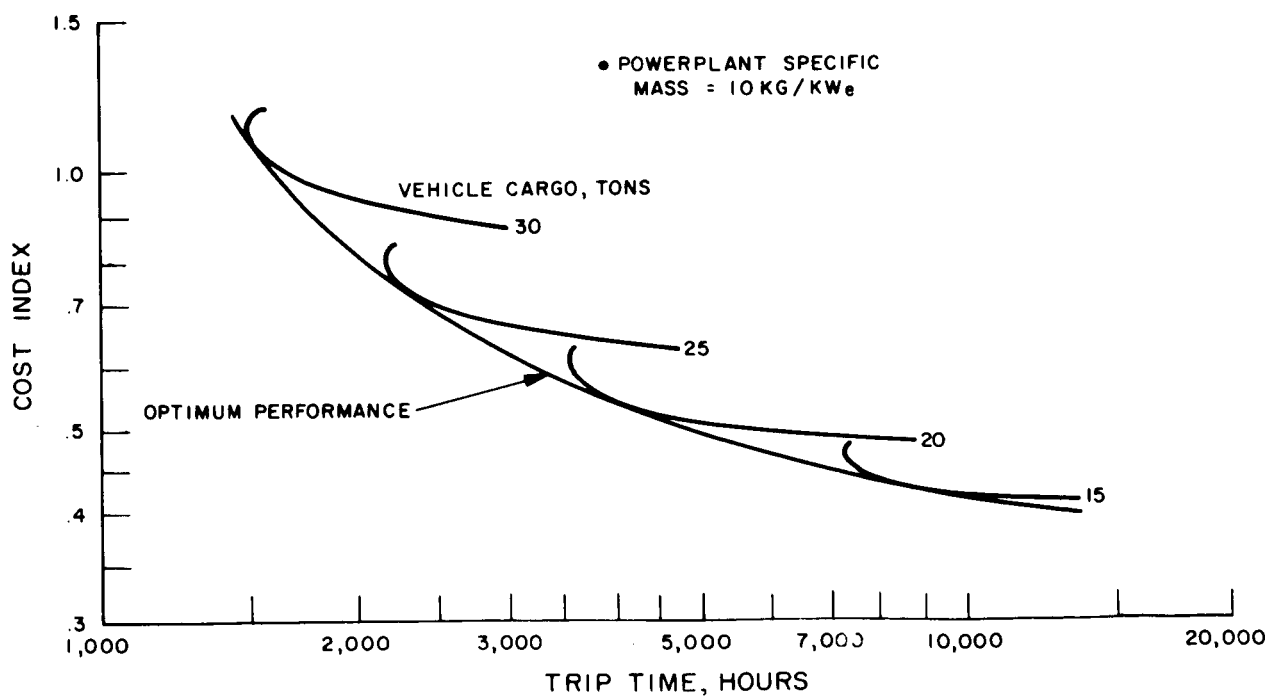
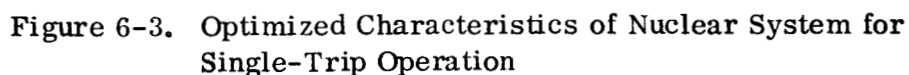


Figure 6-2. Optimization Approach

- The results of using the above optimization method for the single-trip operating mode and nuclear power systems are presented in Figures 6-3 to 6-5. In Figure 6-3, the optimum relationships between vehicle cargo mass, powerplant power, powerplant



specific weight, thruster specific impulse and propulsion time (trip time) are described. The specific impulse is primarily a function of the vehicle cargo mass, and the powerplant power is approximately proportional to the powerplant specific weight. This graph shows that the cargo mass is 2 to 3 times that achievable by the reference chemical rocket system (12,700 kg, or 28,000 lb).

The cost index and power are presented in Figure 6-4 as a function of trip time and powerplant specific weight for the case where the survival penalty and development costs are not considered. These curves indicate that cost is continually decreased with increasing trip time. This conclusion is somewhat erroneous because the probability of component failure and subsequent loss of the lunar cargo is increased with longer trip time. The inclusion of the trip time penalty yields the results presented in Figure 6-5, where a mean time to any failure of 10,000 hours is assumed for the entire cargo vehicle, and redundancy is provided so that the cargo vehicle can survive the first failure. This graph shows, for example, that a 9.06 KG/KW_e (20 lb/kw) powerplant sized at 1.3 MW_e can yield a 30 percent cost savings at a trip time of 4600 hours (six months). The use of a single-trip operating mode determines that the powerplant life required is 4600 hours, which is an easier development goal than the 10,000 hour life requirement usually assumed for nuclear reactor powerplant.

The inclusion of the nuclear powerplant development cost into the mission analysis requires that an assessment be made of the cumulative lunar cargo requirement over the operational life of the transportation concept. The relationship between cost index and cumulative lunar cargo requirement is presented in Figure 6-6 for various trip times. It was estimated in Reference 2 that the cumulative lunar cargo requirement could range between 500 and 2,000 tons over a time span of 15 years. Thus, the acceptance of a 4 to 6 month trip time can yield a 30 to 40 percent cost savings by use of electrical propulsion.

The reuse of the nuclear powerplant in a shuttle type operation can yield a cost savings by the amortization of the nuclear powerplant over a larger total cargo mass. The

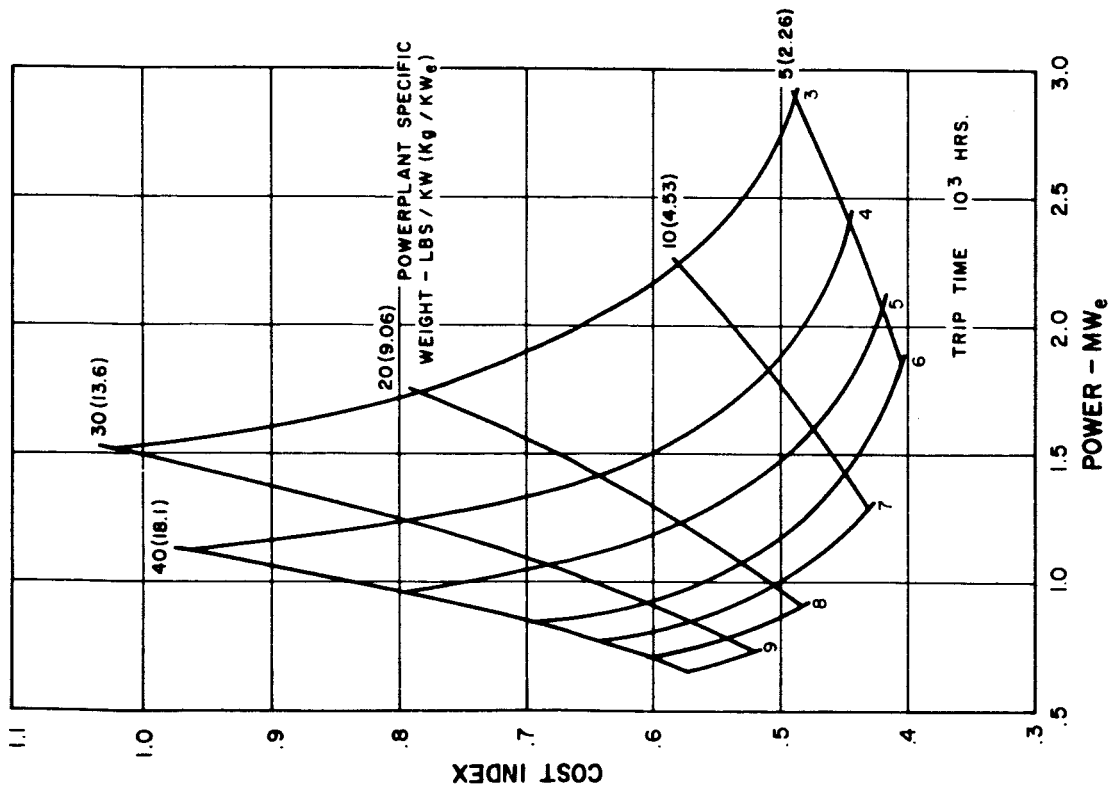


Figure 6-4. Cost Index of Nuclear System for Single-Trip Operation with No Survival Penalty Applied

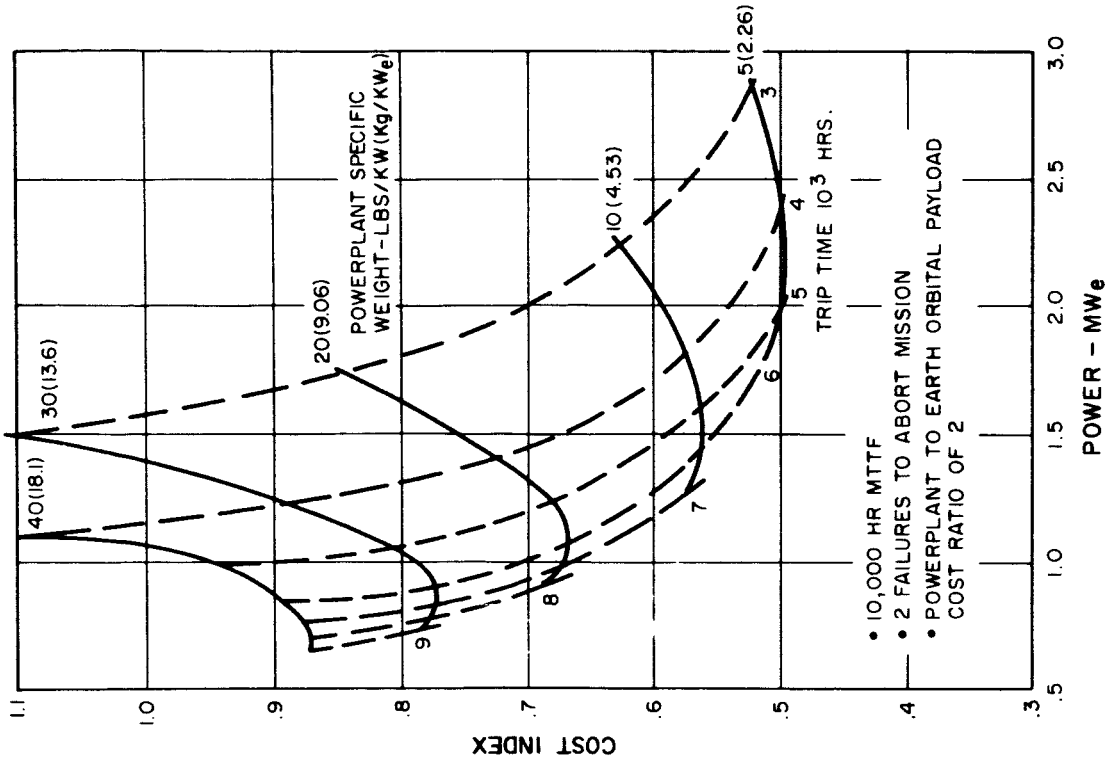


Figure 6-5. Cost Index of Nuclear System for Single-Trip Operation with Survival Penalty Applied

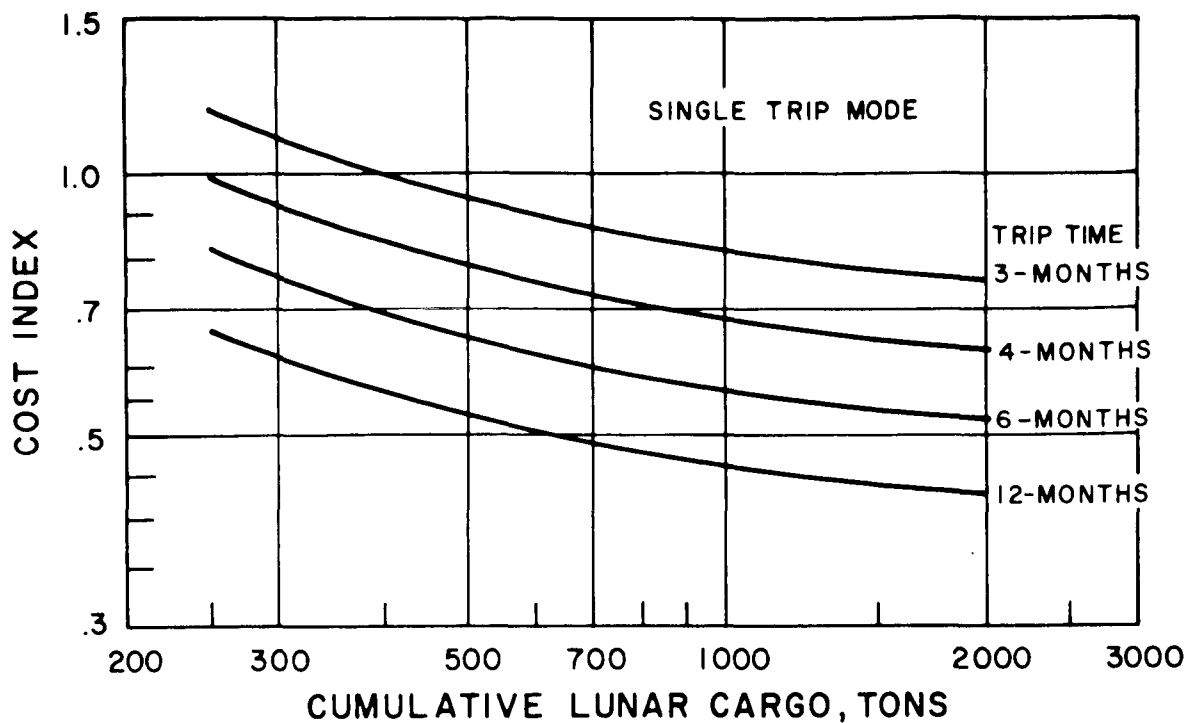


Figure 6-6. Cost Index vs Cargo Requirement of Nuclear System for Single-Trip Operation with Development Cost Considered

effect of the number of trips is shown in Figure 6-7, where cost index is plotted versus time for 1, 2, 3 and 4 trips per powerplant. The solid lines in Figure 6-7 describe the total powerplant operating time requirement, and the dashed lines show the outbound voyage trip times for the 4-trip system. This graph shows that the single-trip mode is always preferable on the basis of powerplant life. However, the 4-trip system is preferable on the basis of trip time. Thus, the shuttle operation provides better performance characteristics than the single-trip operation, but at the cost of several times longer powerplant life capability. A conclusion reached in this study is that the single-trip mode provides the best start for the lunar cargo operations. During the early years of operational service, the powerplant life capability can be expected to improve. When this life is sufficiently long, the operational costs can be further reduced by switching to a shuttle operation. This switch provides an additional advantage in the powerplant disposal area by reducing the number of units involved.

EFFECT OF NUMBER OF TRIPS ON POWER SUPPLY LIFE AND TRIP TIME WITH ION JET THRUSTOR

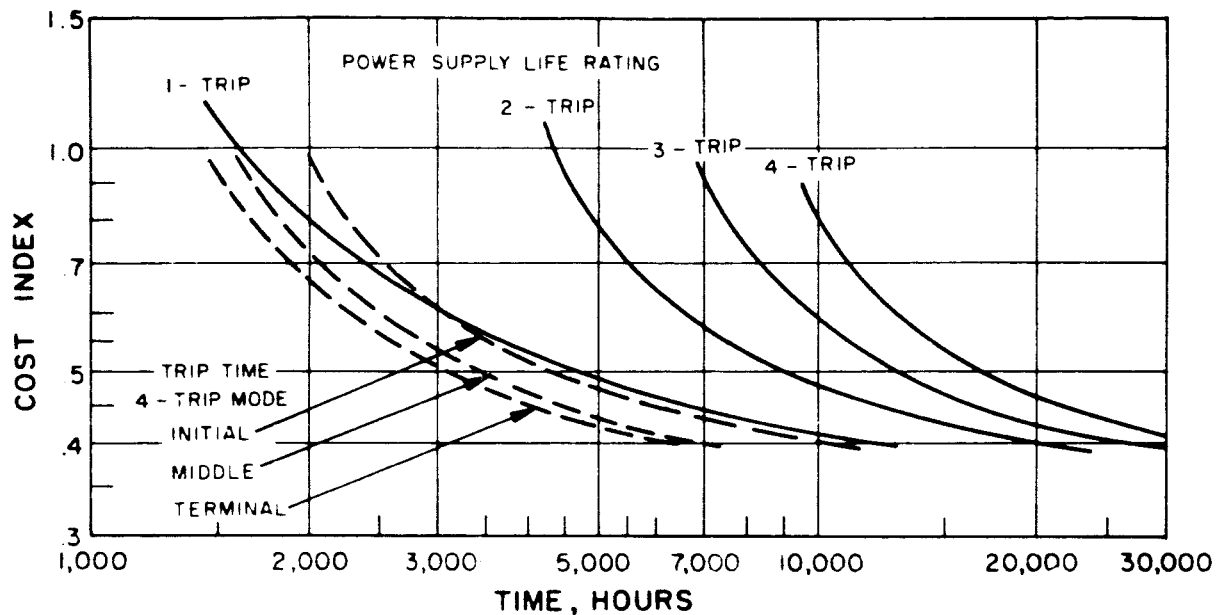


Figure 6-7. Effect of Number of Trips on Nuclear System Cost Index and Trip Time

Further details on the various operational modes examined for shuttle operation are provided in Reference 2.

6.2 NUCLEAR POWERPLANT EVALUATION

A mission study was performed for each of the particular powerplant concepts analyzed in detail and reported in Reference 3. The results of this study are presented below.

The powerplant specific mass versus power relationships developed for beryllium radiators at 5000 hours mean time to puncture were selected for the mission analysis. These specific mass versus power relationships for each of eight powerplant concepts are summarized in Figure 6-8, and they were curve-fitted for use in the mission analysis computer program.

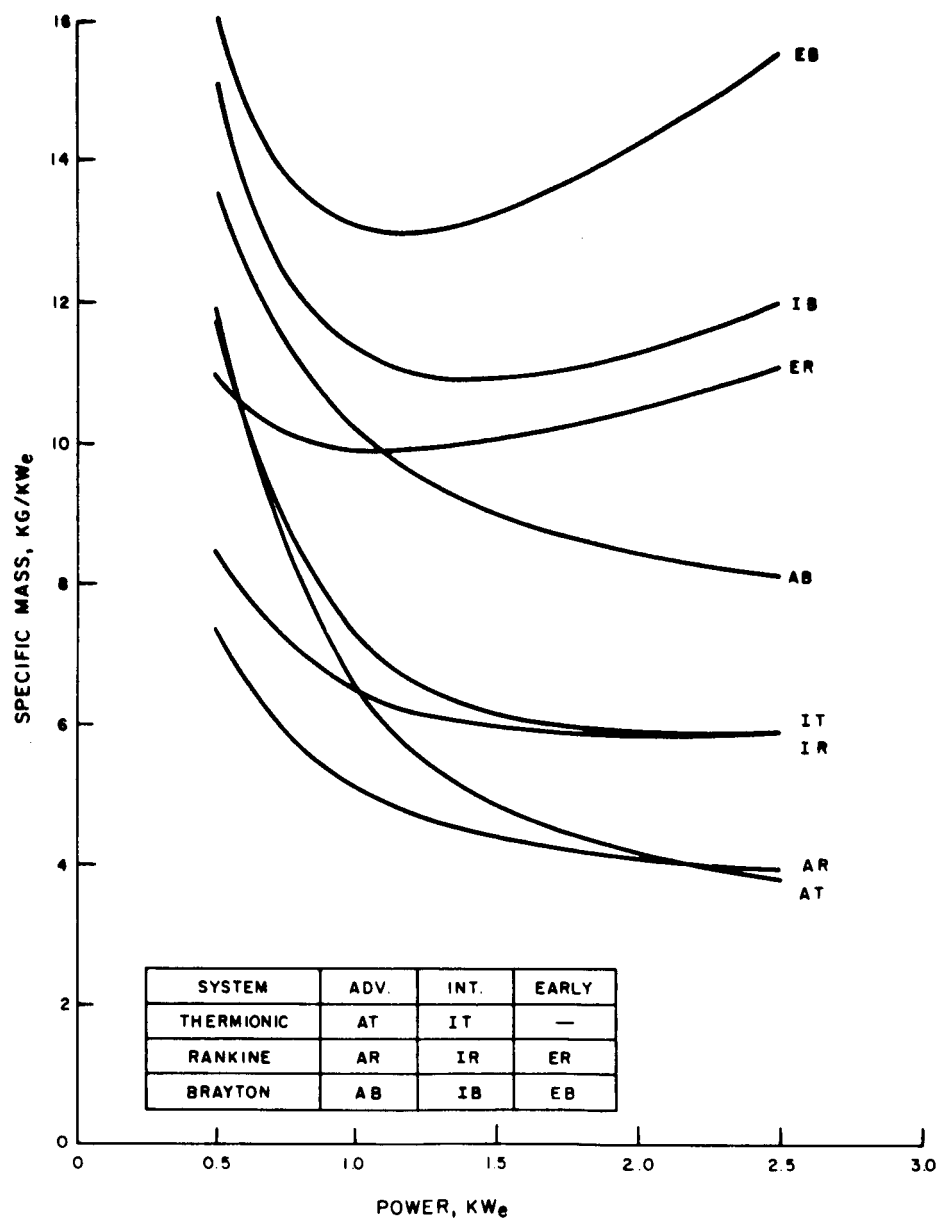


Figure 6-8. Specific Mass vs Power for Various Nuclear Powerplant Concepts

The generalized mission study performed prior to the nuclear powerplant investigation yielded the conclusion that the single-trip operating mode provided favorable mission performance, and that it should be considered for the first generation of electrically-propelled lunar cargo vehicles. On the basis of this conclusion, the nuclear powerplant evaluation was based on the single-trip operating mode, and comparisons in other modes were not made.

The selection criteria developed during the performance of the generalized mission study included the recurring cost for procurement of nuclear powerplants and electrical propulsion systems and the non-recurring cost for the research, development, test and evaluation of the nuclear system. The cumulative lunar cargo requirement was used as an independent variable in a parametric analysis for amortization of the non-recurring cost. The possibility of cargo loss due to powerplant failure was also included in the analysis, and this attrition factor provided a basis for optimization on the basis of minimum cost. Using this approach the cargo system optimized at rather long trip times. However, the various non-optimum cost versus trip time characteristics were also determined and prepared in a parametric presentation. The worth of trip time in terms of cost is not readily determined and was ignored during the generalized mission study. This approach makes a powerplant comparison rather complex. To facilitate the comparison of the various powerplant concepts, a trip time worth, or cost versus trip time tradeoff, has been added to the mission analysis.

Provision for a tradeoff between cost and trip time has been added to the mission analysis program by use of the independent parameter, "Time Worth Index", defined by the following equation:

$$C_A = C (1 + C_T T)$$

where

C = Cargo Deliver Cost, using nuclear-electric system

C_A = Apparent Cost

C_T = Time Worth Index

T = Trip Time, yr

The Time Worth Index provides a correction to cargo delivery cost, which represents a penalty that is proportional to trip time. Then, the nuclear-electric system parameters are optimized on the basis of minimum apparent cost. This approach provides

a basis for consistency in selecting data points for comparisons between the various nuclear powerplant concepts. To illustrate the significance of the Time Worth Index, consider the following case. A value of 0.5 for Time Worth Index causes the apparent cargo delivery cost to increase by 50 percent for a trip time of one year. Thus, the situation wherein an electrically-propelled cargo vehicle provided a 33 percent cost reduction over chemical systems would be a standoff at a trip time of one year. The selection of a value for Time Worth Index is somewhat arbitrary, and 0.5 has been assumed for performing the nuclear powerplant comparison.

The cost index used for the powerplant optimization is presented below:

$$C_I = (1 + C_P + C_D W_N/W_T) (W_C/W_N)$$

where

C_I = Cost Index = ratio of electrical propulsion system to reference chemical rocket system cargo delivery cost

C_P = (Powerplant Cost)/(Booster Cost) = Nuclear Powerplant Cost Index

C_D = (Development Cost)/(Booster Cost) = Development Cost Index

W_C = Cargo mass delivery of reference chemical rocket system

W_N = Vehicle Cargo, cargo mass delivery of electrically-propelled lunar cargo vehicle

W_T = Cargo Requirement, cumulative or total cargo to be delivered to the lunar surface over the service life of the powerplant concept.

This cost index is then multiplied by $(1 + C_T T)$, where C_T is the Time Worth Index and T is the trip time as previously discussed, yielding an apparent cost index to be minimized for selection of power and specific impulse.

The powerplant procurement cost is related to the powerplant mass, power level, and concept selection. In the present study, the cost has been assumed to be proportional to mass alone, and the other factors have been neglected. The constant of proportionality is defined as Nuclear Powerplant Cost Index, which is equal to powerplant

cost in dollars per metric ton divided by booster cost. The relationship between Nuclear Powerplant Cost Index, powerplant cost and powerplant mass is presented in Figure 6-9. On the basis of this graph, a 10 ton powerplant constructed at \$4000/kg (\$1817/lb) mounted on top of a \$100 million booster cost \$40 million to procure, and has a Nuclear Powerplant Cost Index of 0.04.

The powerplant development cost is even more involved than the procurement cost. This cost has been assumed to be a function of powerplant mass and qualified operating time (trip time) alone in the present study. The assumed function is presented in Figure 6-10.

These assumed cost relationships are based on rather preliminary unpublished work performed in relationship to the SNAP-50 program. The inclusion of these costs has greatly benefitted the mission study, even though the cost assumptions are rather crude and approximate.

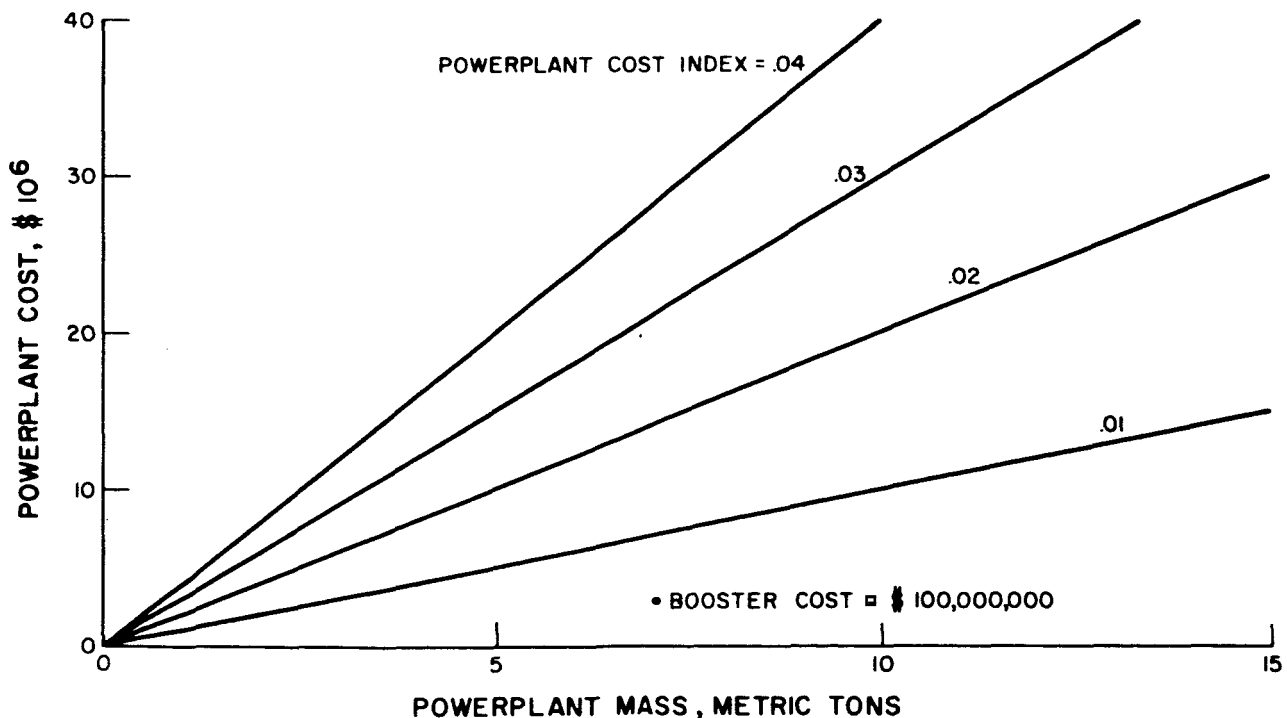


Figure 6-9. Powerplant Cost Index as a Function of Mass and Procurement Cost

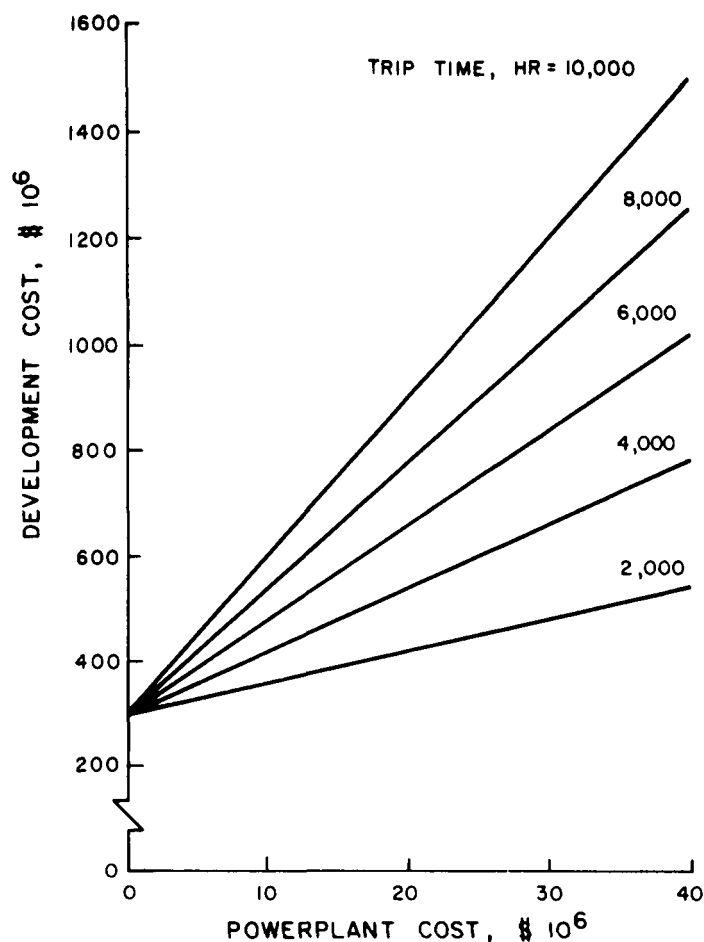


Figure 6-10. Assumed Powerplant Development Cost Relationship

The results of performing the mission optimization for one set of cost factors are presented in Figure 6-11 to 6-18 for each of the powerplant concepts. In this optimization the powerplant size and thruster specific impulse were varied to minimize the cost index as defined by the equation above. The resulting characteristics were determined for a range of values for Nuclear Powerplant Cost Index. These data are based on 0.5 Time Worth Index, and 2000 metric tons total cargo requirement.

The data in Figures 6-11 to 6-18 include variations of cost index, vehicle cargo, power, trip time, and specific impulse with Nuclear Powerplant Cost Index. All of these parameters except vehicle cargo show sizable variations. The cost index varies by a range of 0.18 in magnitude for the range of data shown. The trip time varies by 1000 hours; the specific impulse, by 1500 seconds; and the power, by 0.6 MW.

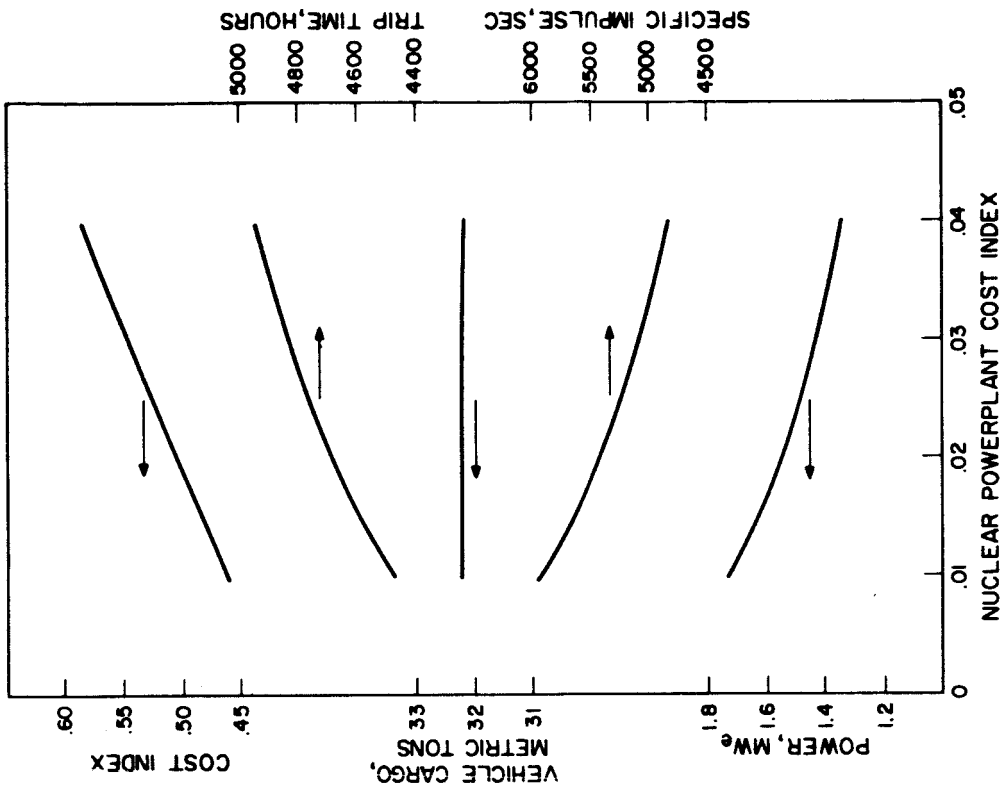


Figure 6-11. Advanced Technology Thermionic System Optimized Characteristics

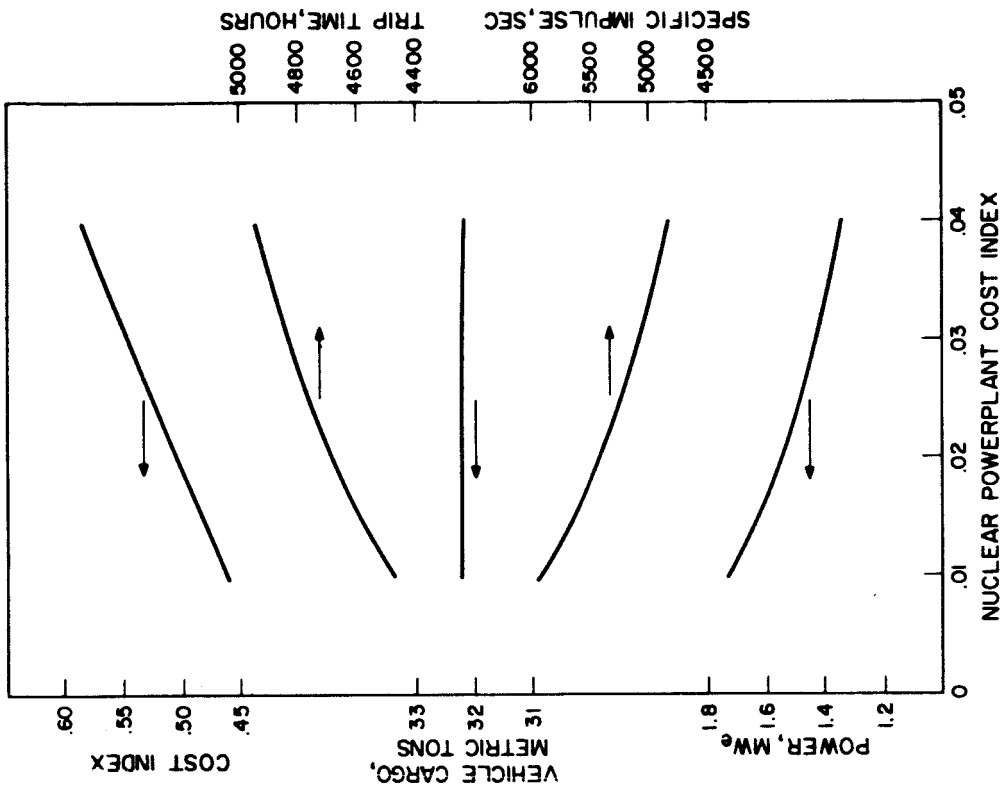


Figure 6-12. Intermediate Technology Thermionic System Optimized Characteristics

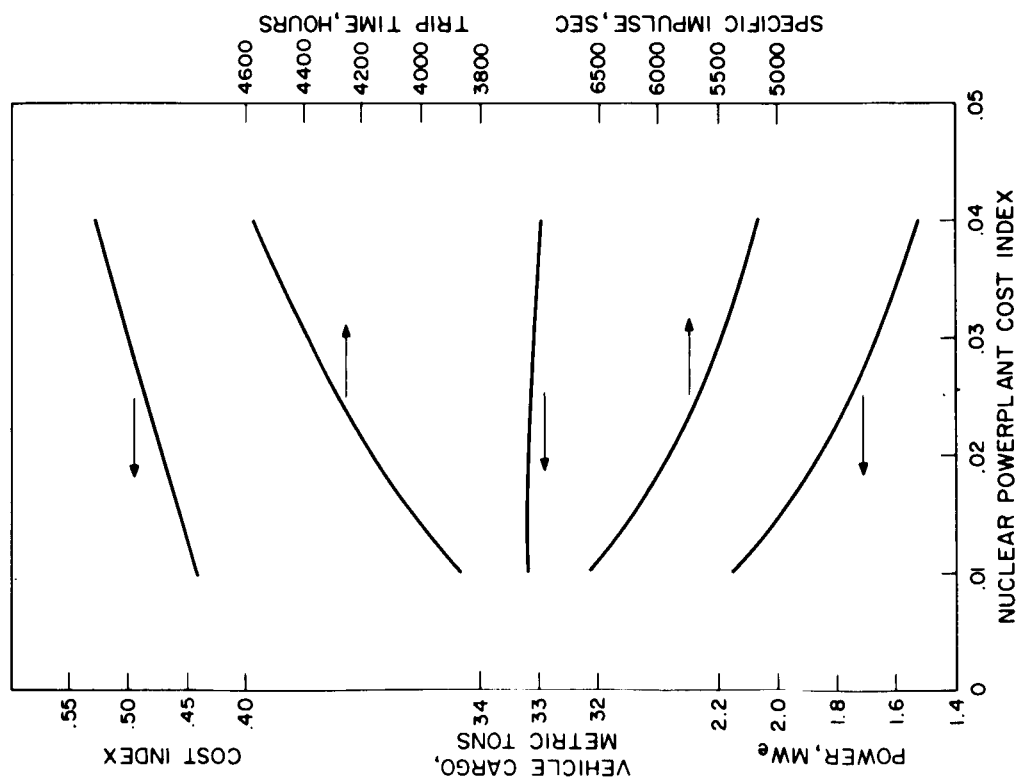


Figure 6-13. Advanced Technology Rankine System Optimized Characteristics

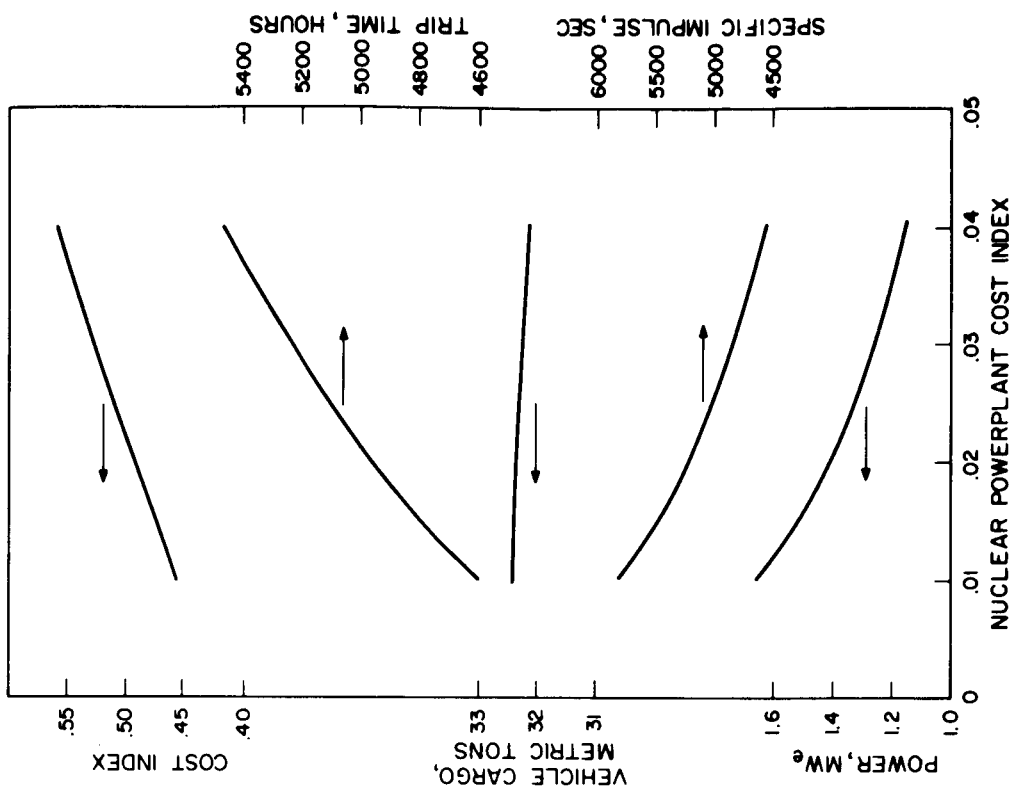


Figure 6-14. Intermediate Technology Rankine System Optimized Characteristics

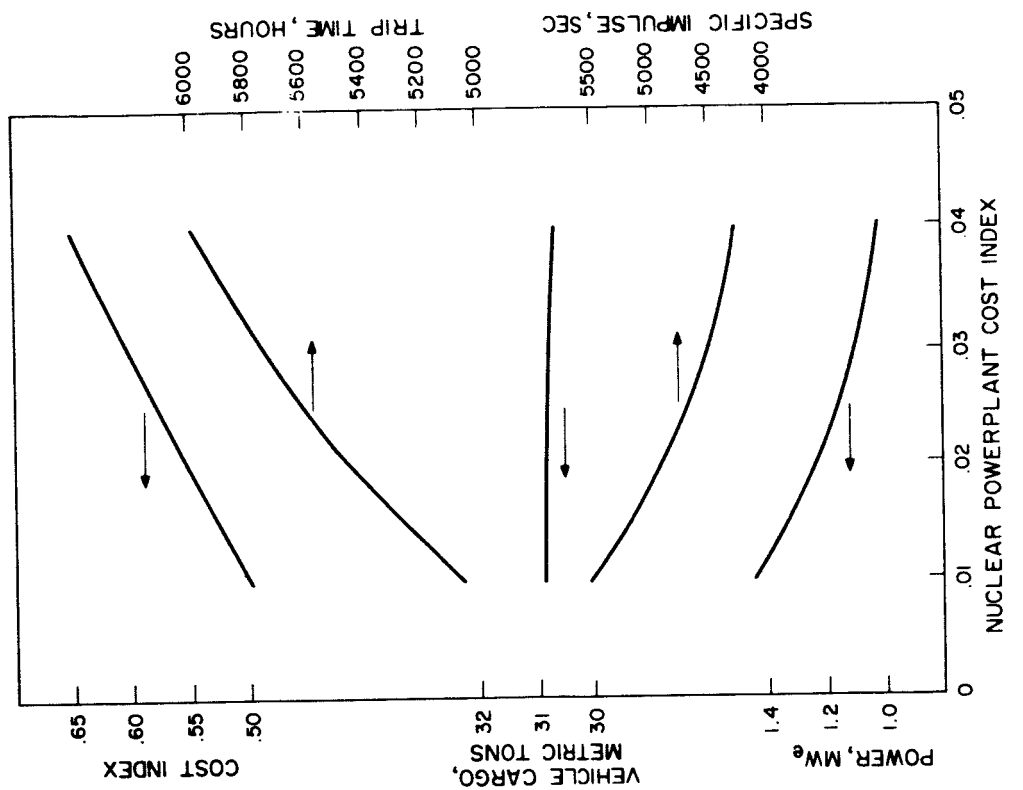


Figure 6-15. Early Technology Rankine System Optimized Characteristics

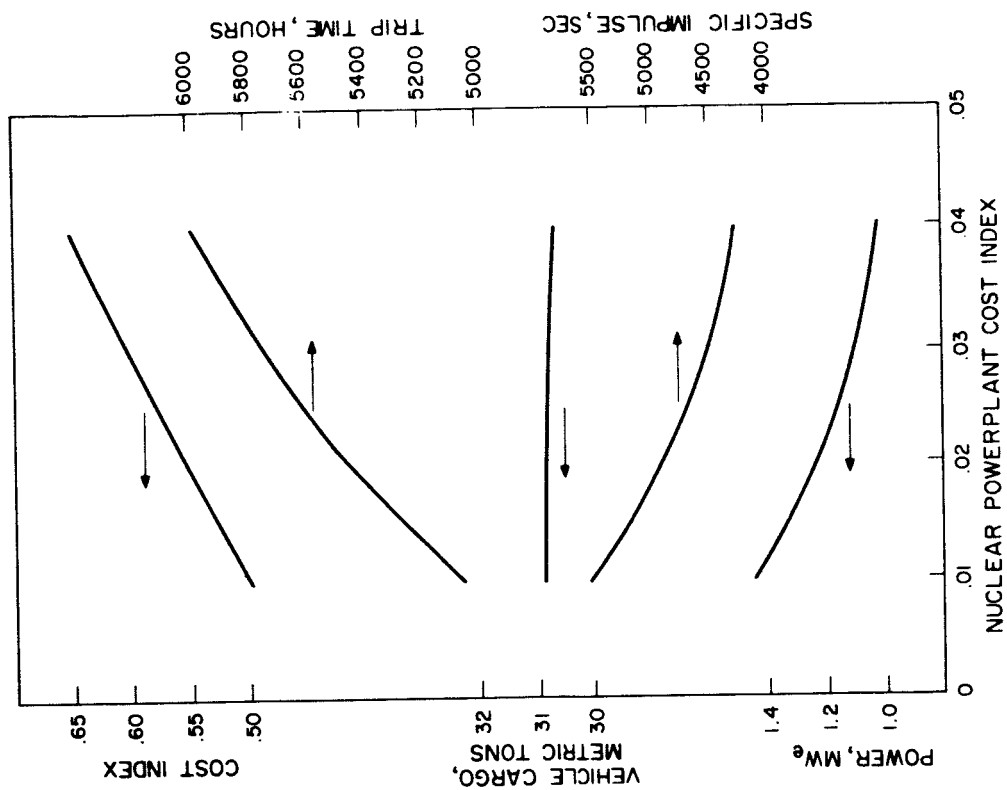


Figure 6-16. Advanced Technology Brayton System Optimized Characteristics

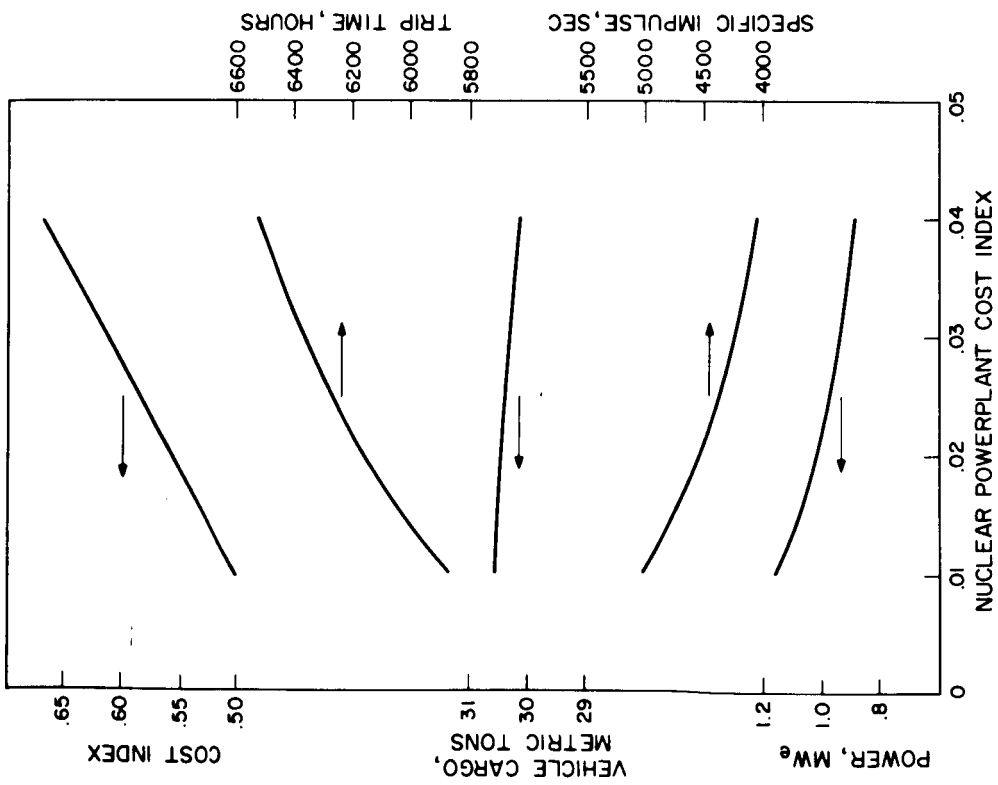


Figure 6-17. Intermediate Technology Brayton System Optimized Characteristics

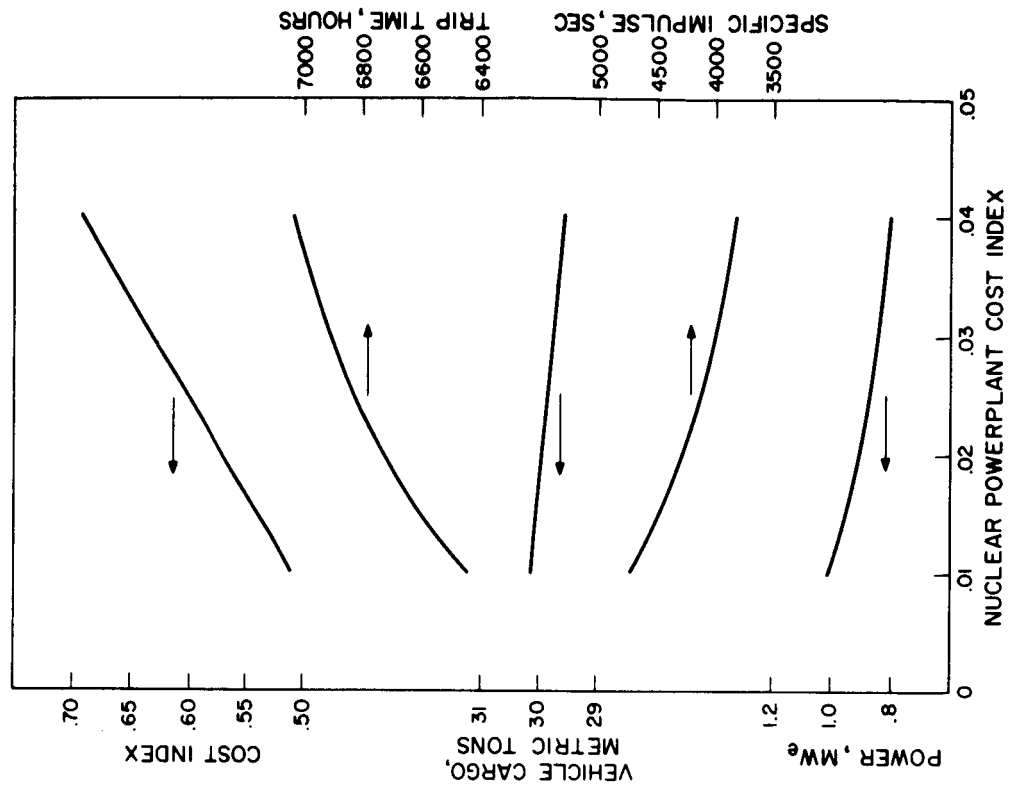


Figure 6-18. Early Technology Brayton System Optimized Characteristics

The comparison of mission performance for the eight nuclear powerplant concepts is presented in Figure 6-19. The trip time ranges between 3500 and 7100 hours; the cost index, between 0.44 and 0.69. This graph shows that Nuclear Powerplant Cost Index is a very significant factor. On the basis of cost, the advanced technology Rankine system has the lead. The advanced thermionic system provides a shorter trip time, but at slightly greater cost. It is difficult and somewhat misleading to draw conclusions regarding powerplant selection from this graph. The appropriate Nuclear Powerplant Cost Index can be expected to vary from one powerplant concept to another. Also, the development cost model assumed for each of the eight concepts is identical. The conclusion that can reasonably well be drawn, however, is that several of the powerplant concepts assumed do provide attractive cost savings for the lunar cargo operation. The trip time will tend to be more dependent on the powerplant concept than the cost; the powerplant cost will influence primarily the cargo delivery cost.

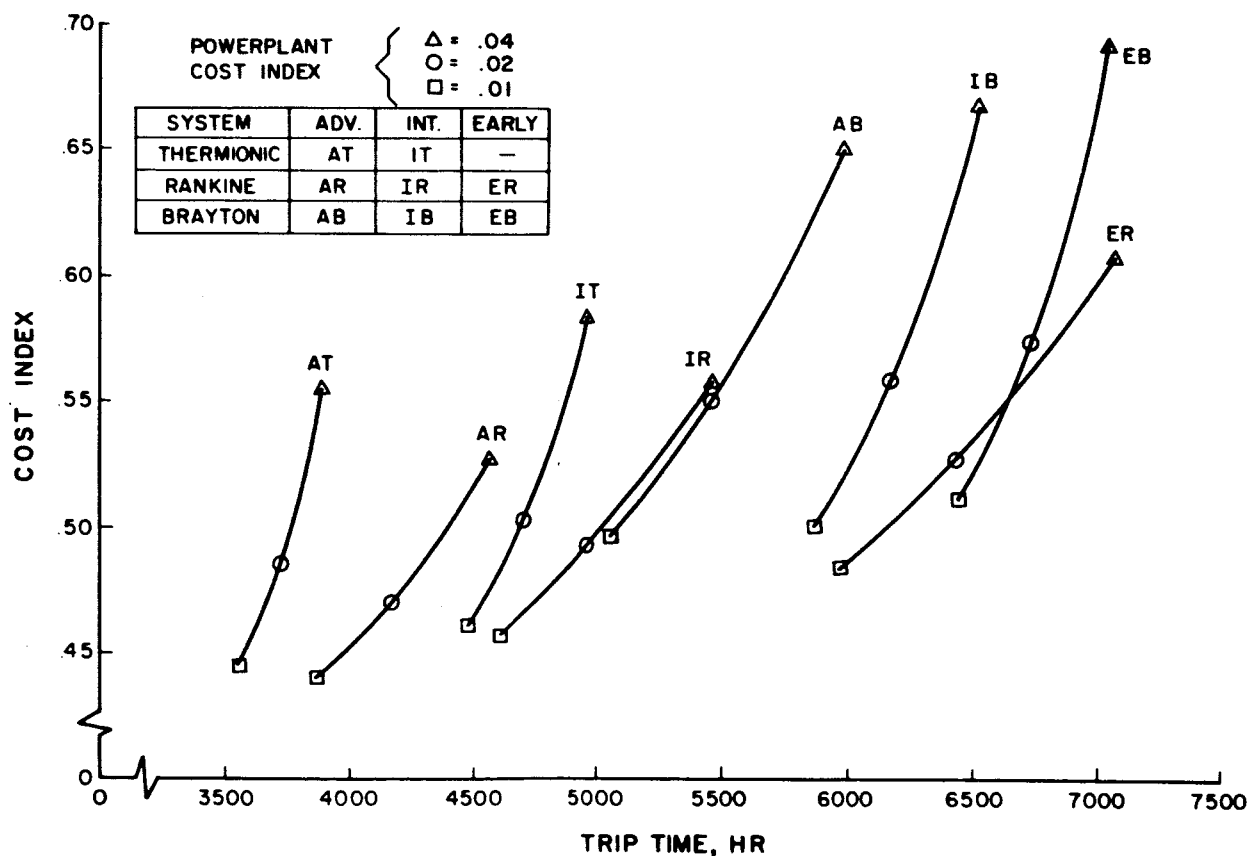


Figure 6-19. Comparison of Cost and Trip Time for Various Nuclear Powerplant Concepts

The effect of the Time Worth Index selection on the powerplant characteristics is shown in Figures 6-20 to 6-22, wherein the Nuclear Powerplant Cost Index is 0.02. Results are presented for the advanced technology thermionic system, the intermediate technology Rankine system, and the early technology Brayton system. These graphs describe the envelope of trip time, cost index, vehicle cargo, power and specific impulse to be considered for the single-trip operating mode in the electrically-propelled lunar cargo vehicle. Figure 6-20 shows that the trip time can be shortened by about 40 percent for any of the powerplant concepts and still show a cost savings if other factors made shortened trip time even more important than the trip time penalty assumed. The data in Figure 6-20 does indicate that the use of a constant Time Worth Index for powerplant comparisons can be expected to provide reasonably good consistency for powerplant comparisons.

The vehicle cargo mass per trip is shown in Figure 6-21 to range between 24 and 35 metric tons. These values are premised on the basis of 0.4 cargo fraction for the lunar landing vehicle. This large size lander eases the problem of insulation for the lander cryogenic propellant tanks, and values up to 0.45 have been estimated. (See Reference 2). At a Time Worth Index of 0.5, the vehicle cargo mass variation is between 30 and 33 tons. Thus, the lander gross mass is between 75 and 82 tons, and the size of lunar landing vehicle is narrowly bracketed by this study.

Figure 6-22 shows the variation of specific impulse and electrical power. The envelope variations for the powerplant concepts is substantial, which is from 0.75 to 3.0 MW_e on power, and 2500 to 7500 seconds on specific impulse. The 0.5 Time Worth Index narrows the power and specific impulse variations about in half. The power ranges between 1 and 2 MW_e; the specific impulse, between 4300 and 6400 seconds. These selections will remain to be in question until the choice of powerplant concept has been further narrowed.

The effect of varying the cargo requirement as well as the Nuclear Powerplant Cost Index on the optimum characteristics for the advanced technology thermionic system is presented in Figures 6-23 to 6-25. In Figure 6-23, cost index varies from 0.45 to 0.70; the trip time, from 3100 to 3900 hours. The cargo requirement is shown to have a sizeable effect on trip time, and the smaller cargo requirement leads to shorter optimum trips at higher cost. In Figure 6-24, the optimum vehicle cargo is shown to decrease slightly with variation of cargo requirement. The optimum power and specific impulse is presented in Figure 6-25, which shows the power to increase and the specific impulse to decrease with decreasing cargo requirement.

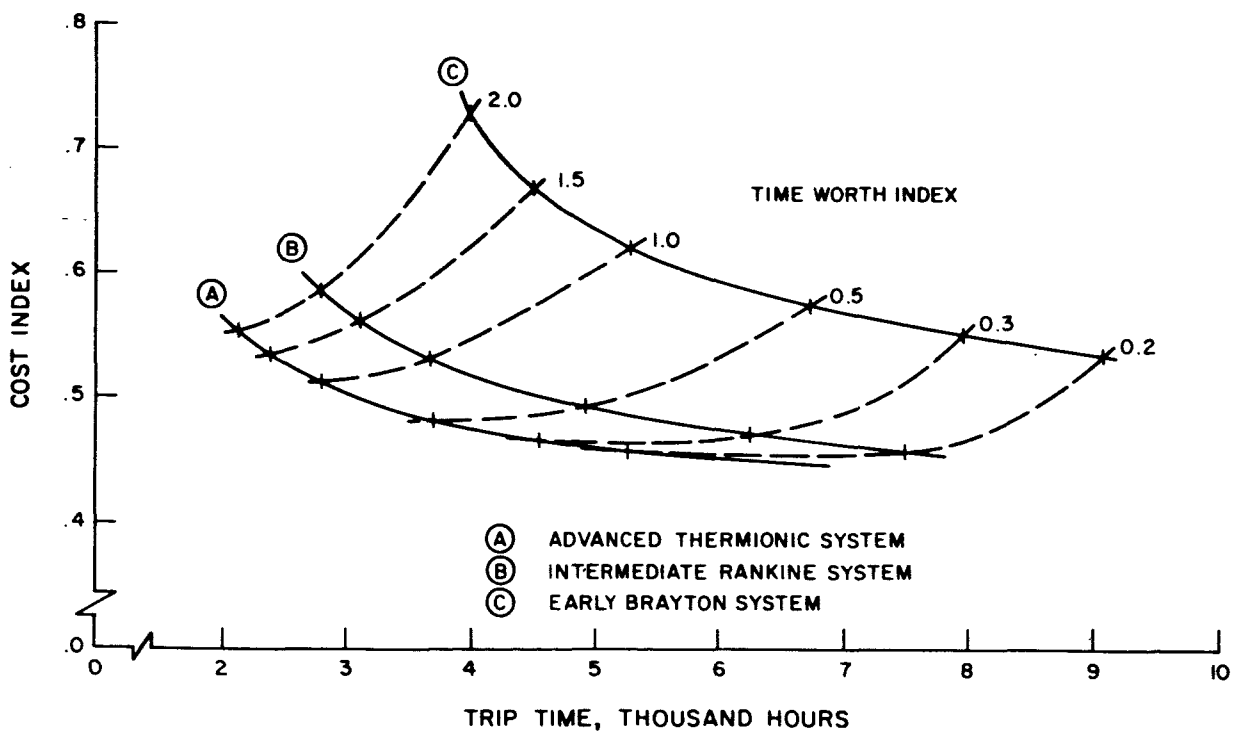


Figure 6-20. Cost Index vs Trip Time Variation with Time Worth Index

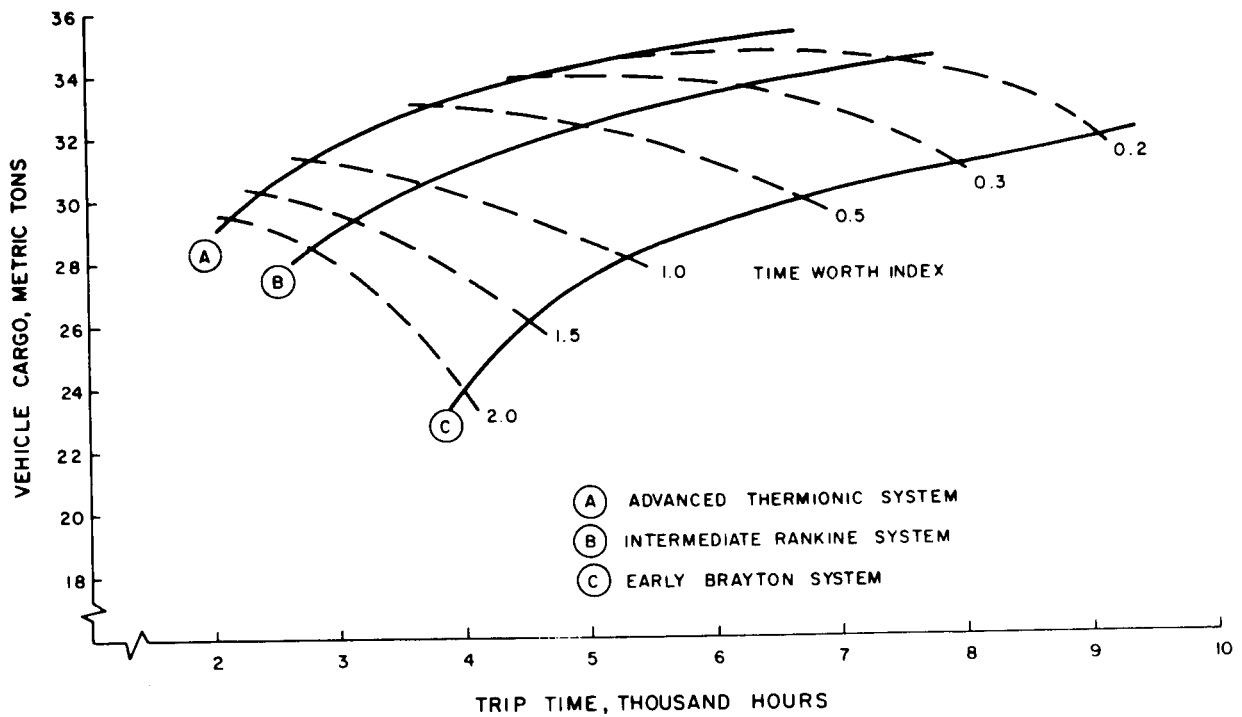


Figure 6-21. Cargo Tonnage vs Trip Time Variation with Time Worth Index

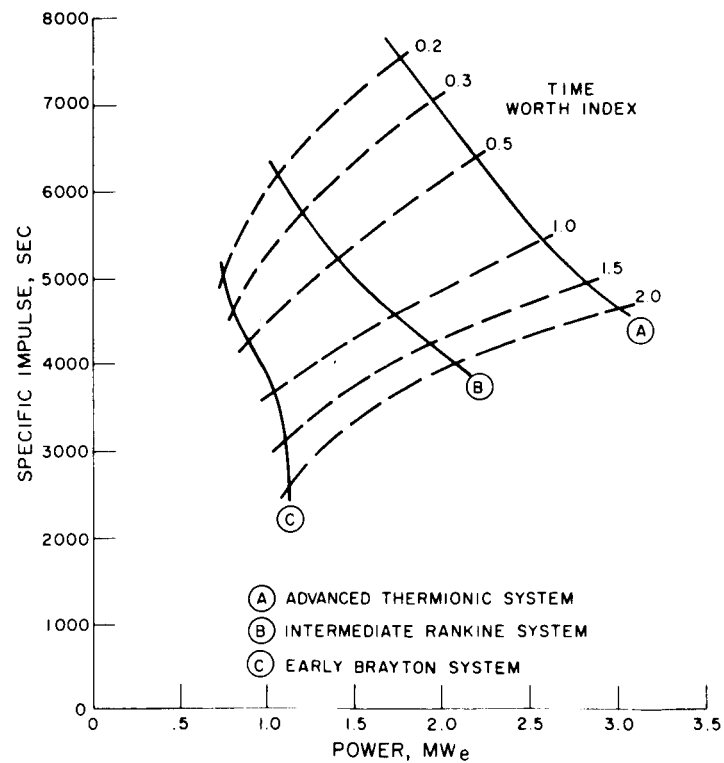


Figure 6-22. Optimum Power and Specific Impulse Variation with Time Worth Index

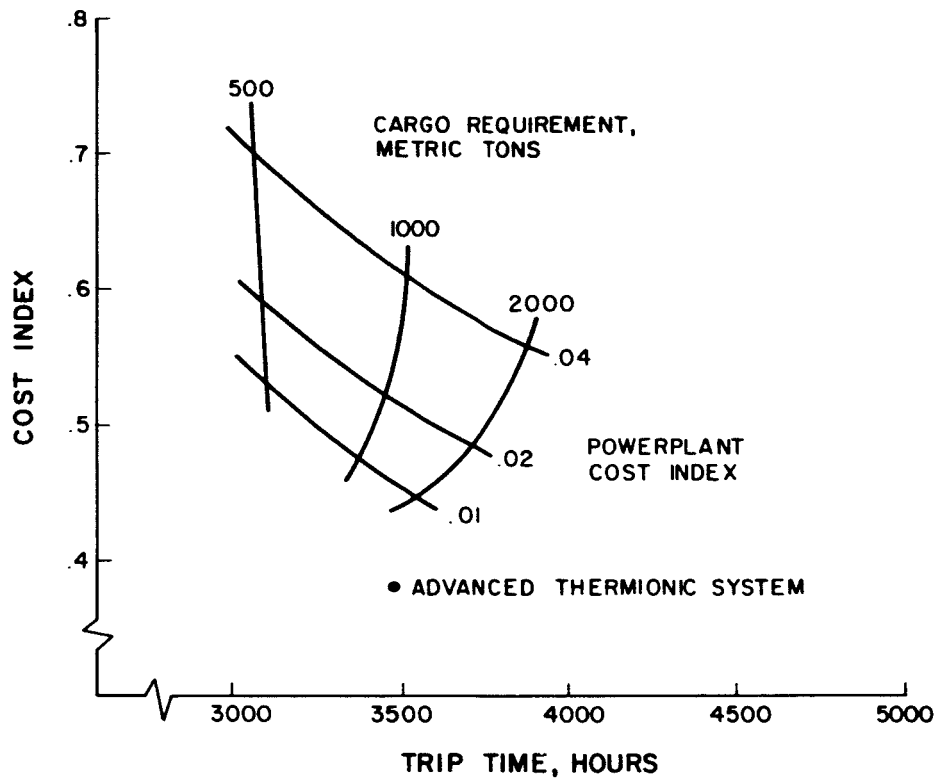


Figure 6-23. Cost Index vs Trip Time Variation with Cumulative Lunar Cargo

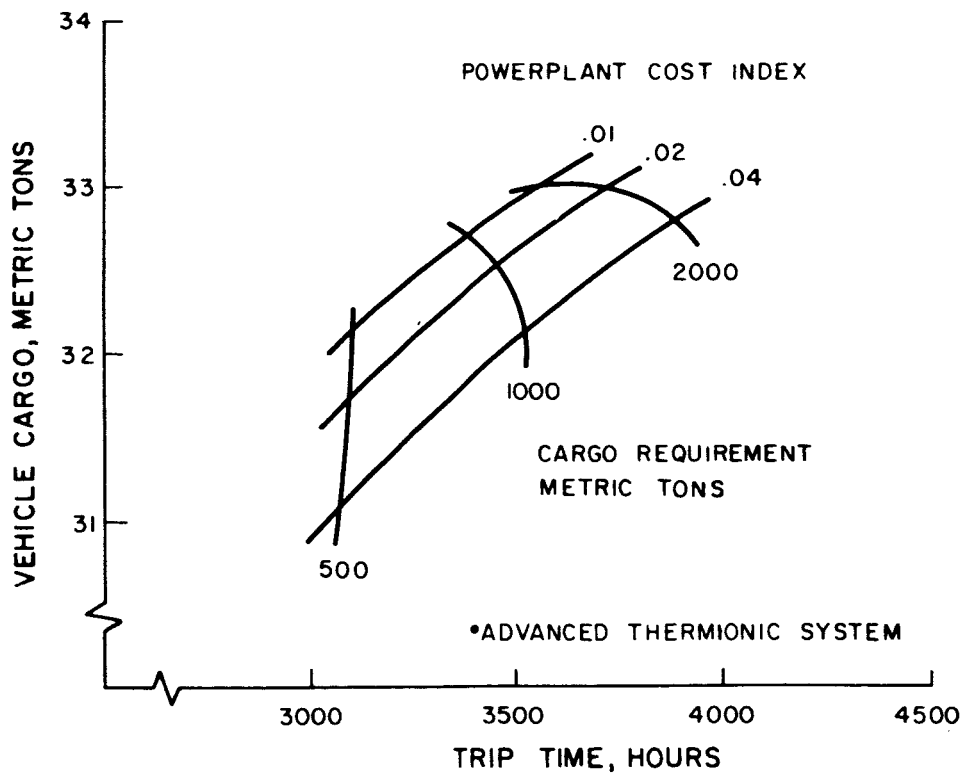


Figure 6-24. Lunar Vehicle Cargo vs Trip Time Variation with Cumulative Lunar Cargo

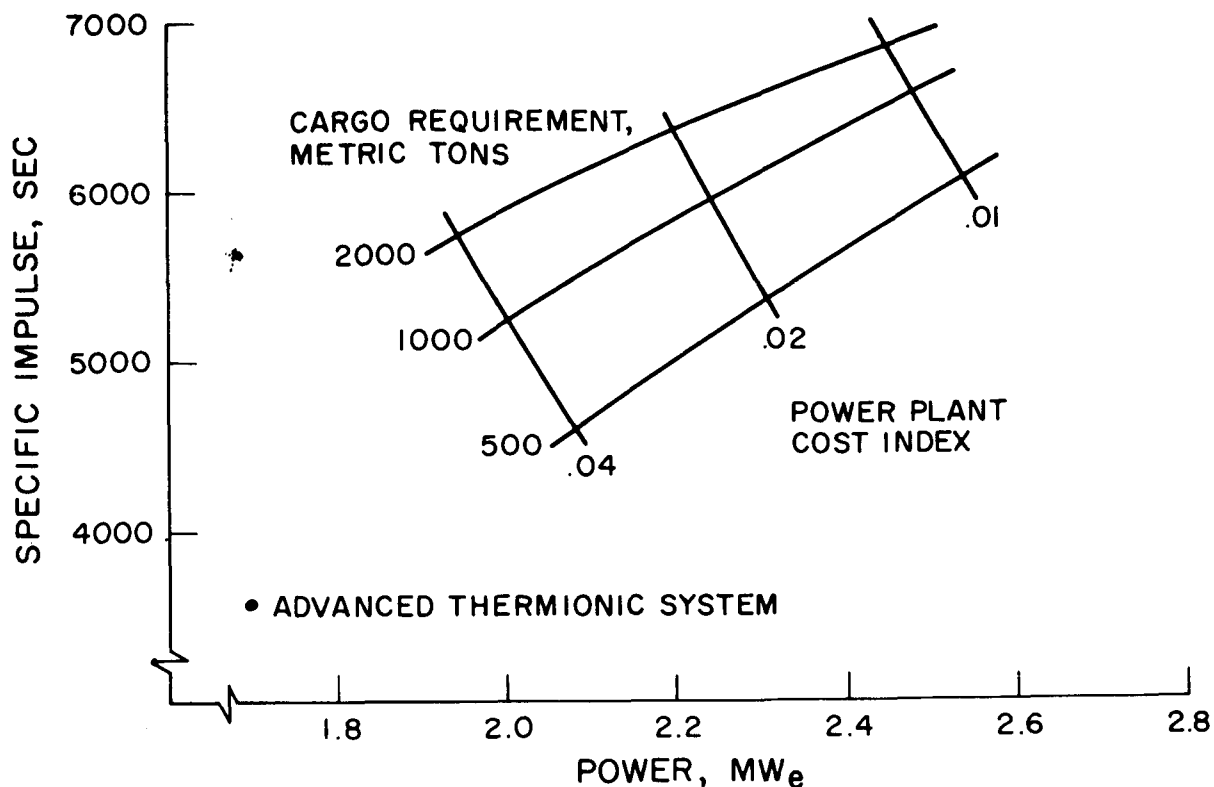


Figure 6-25. Optimum Power and Specific Impulse Variation with Cumulative Lunar Cargo

6.3 SOLAR POWERPLANT EVALUATION

The mission analysis of an electrically-propelled cargo vehicle employing a solar power source is readily performed following determination of values for the items below:

- Characteristic Velocity (for propulsion requirements)
- Shadow time fraction (for trip time estimation)
- Powerplant mass in kilograms per unit of power
- Powerplant cost in dollars per unit of power

The trajectory analysis performed during this study determined that the characteristic velocity for solar power systems is three percent larger than that for nuclear power systems due to the earth shadow effect. This earth shadow also caused the trip time to increase by 18 percent. However, the relationships between powerplant mass, cost and power can not be accurately determined, and they are treated parametrically in this mission study.

The powerplant specific mass for a silicon solar cell power supply system used in the recent Mariner IV spacecraft flight is 45 KG/KW_e (100 LB/KW_e). Recent studies and experimental panel fabrications have shown promise for 22 KG/KW_e (50 LB/KW_e) in 50 KW_e size arrays. The possibility has been shown in analytical studies that 11 KG/KW_e (25 LB/KW_e) can be achieved in the near future. Based on these factors, a range of specific mass between 10 and 30 KG/KW_e has been selected for the parametric mission analysis.

The solar array cost is currently estimated at the magnitude of \$500 per watt. On this basis a one megawatt array would cost \$500 million, or five times the cost of a Saturn V. The cost would have to decrease by more than a factor of 10 to be competitive with chemical rocket systems in a single-trip operation. However, the solar system is inherently long lived and an operational life for a solar array of five to ten years can be envisioned. On this basis a shuttle operational mode was selected for use in the solar system evaluation. The cost of the solar array was treated parametrically on the basis of cost per unit of power per unit of time. A Solar Array Cost Index is defined, which is cost per megawatt of electrical power per year of operation divided by the booster cost. A range of Solar Array Cost Index between 0.2 and 0.6 was used in the parametric mission analysis. The Solar Array Cost Index of 0.2, for example, could correspond to the situation wherein a one megawatt array is constructed and placed into orbit at a cost of \$200 per watt, operates for ten years, and transports lunar landing vehicles launched from earth on \$100 million boosters to lunar orbit.

The general performance characteristics of the solar powered, electrically propelled, lunar cargo shuttle are presented in Figures 6-26 and 6-27 for solar array specific masses of 10 and 20 KG/KW_e, respectively. These curves show that at a specific mass of 10 KG/KW_e the trip time can be as low as 2500 hours using a vehicle cargo size of 20 metric tons and a 3 megawatt solar array. Also, the vehicle cargo can be greater than 35 metric tons and much smaller size solar arrays can be employed if the trip time is ignored. The inclusion of costing considerations serves to narrow the scope of the parameters of interest.

The approach described for the evaluation of nuclear powerplants in the previous section, wherein a cost index was determined and a Time Worth Index was applied, has been applied in a similar manner to the solar powerplant mission evaluation. On the basis of 0.5 Time Worth Index, the variation of system characteristics with Solar Array Cost Index and solar array specific mass have been determined and are presented in Figures 6-28 to 6-30. The solar array specific mass is shown to have a major effect on trip time in Figure 6-28. This trip time doubles in magnitude over the range of variables used. The Solar Array Cost Index has strong effect on cost index, and values of Solar Array Cost Index above 0.6 show little advantage for electrical propulsion over the reference chemical rocket system.

The vehicle cargo is determined from Figure 6-29 to be between 29 and 32 metric tons, which is a rather narrow range and is similar to the results shown for the nuclear

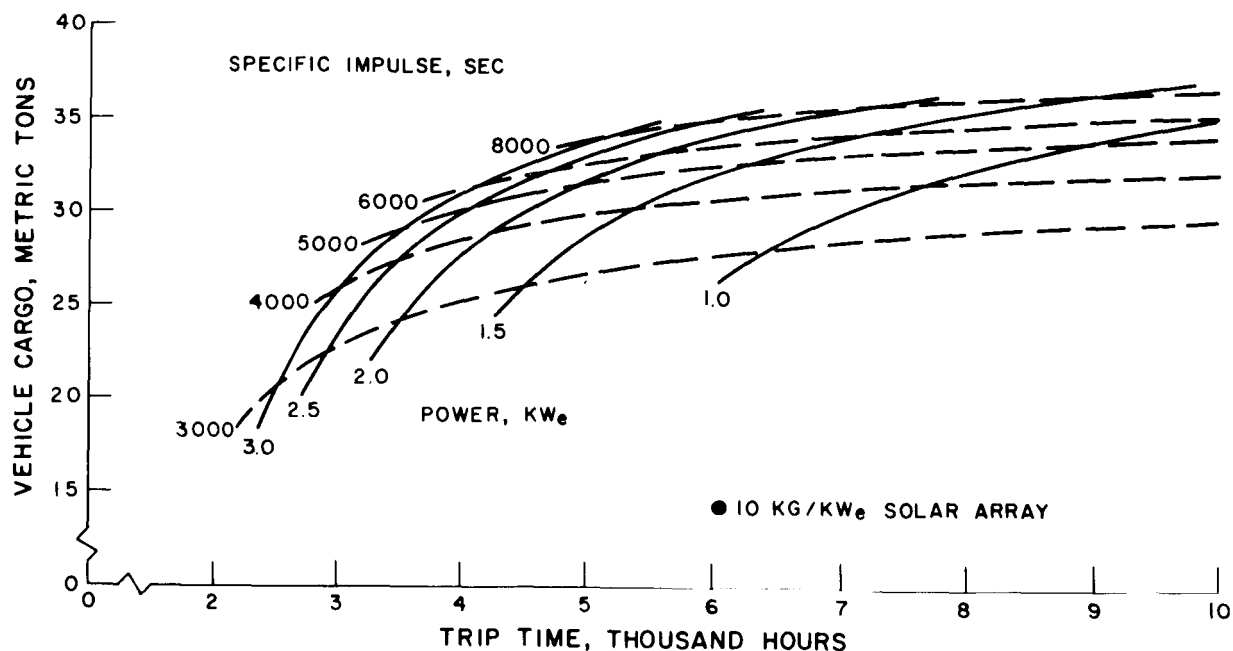


Figure 6-26. General Performance Characteristics of Solar System at 10 KG/KW_e for Shuttle Operation

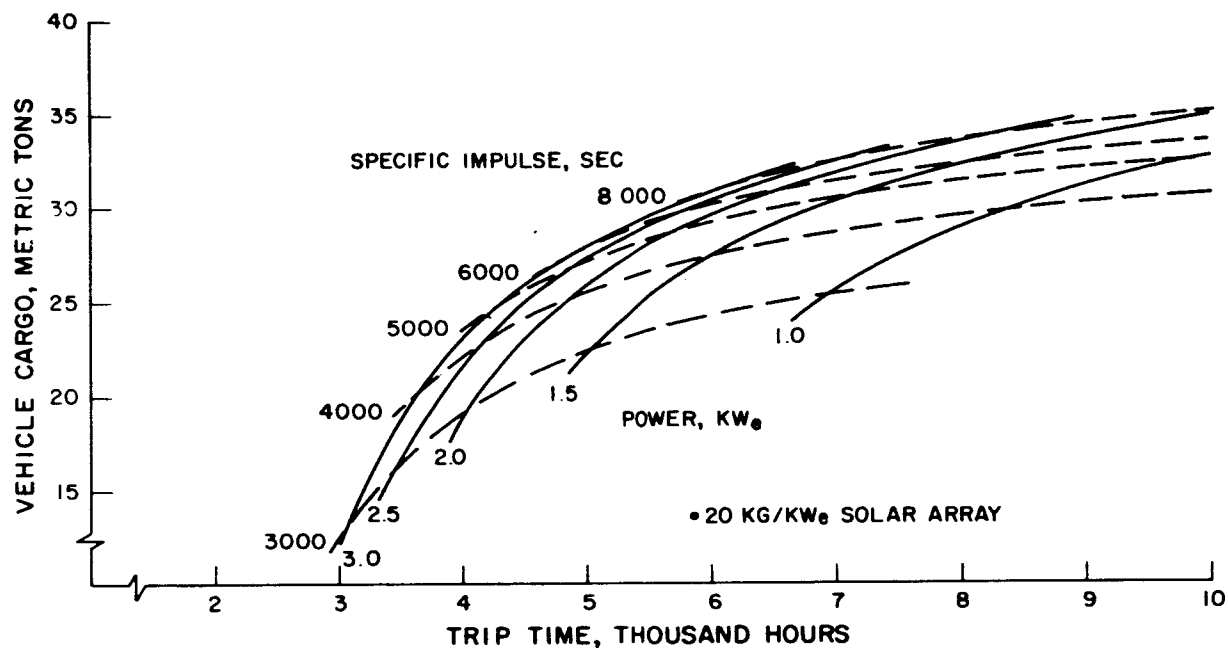


Figure 6-27. General Performance Characteristics of Solar System at 20 KG/KW_e for Shuttle Operation

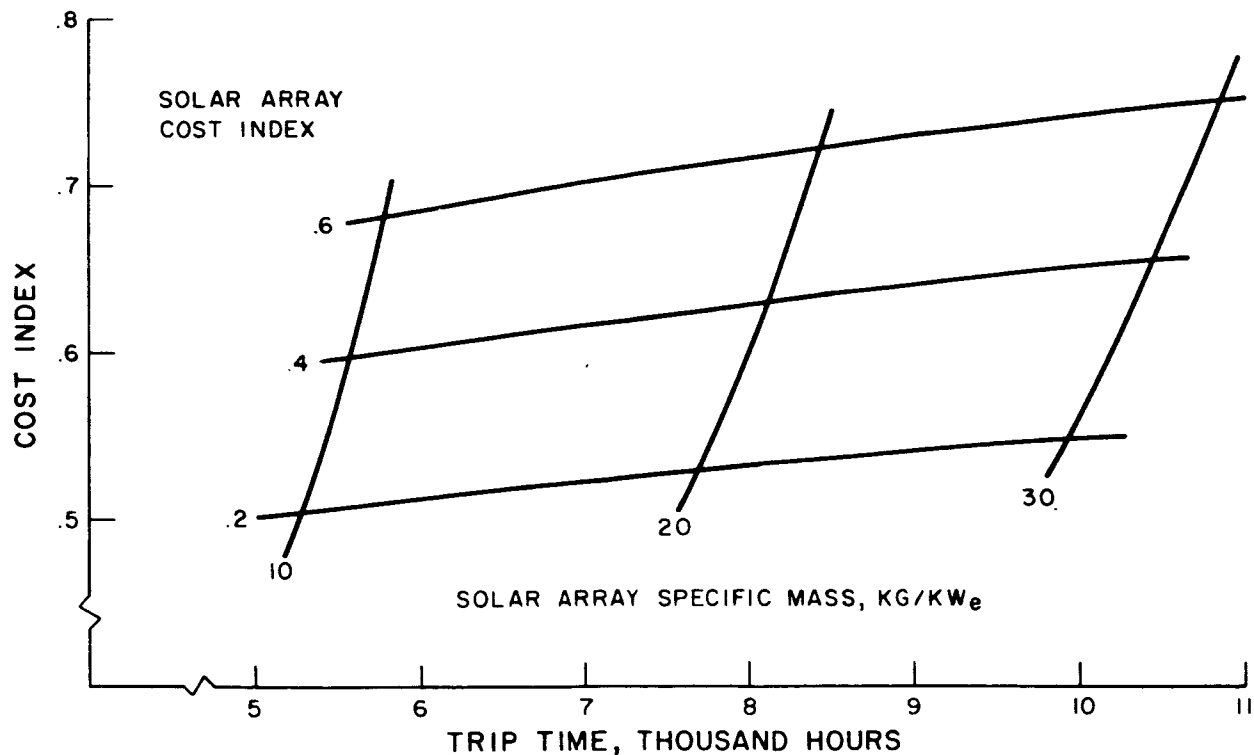


Figure 6-28. Cost Index and Trip Time Variation with Solar Array Cost Index and Specific Mass

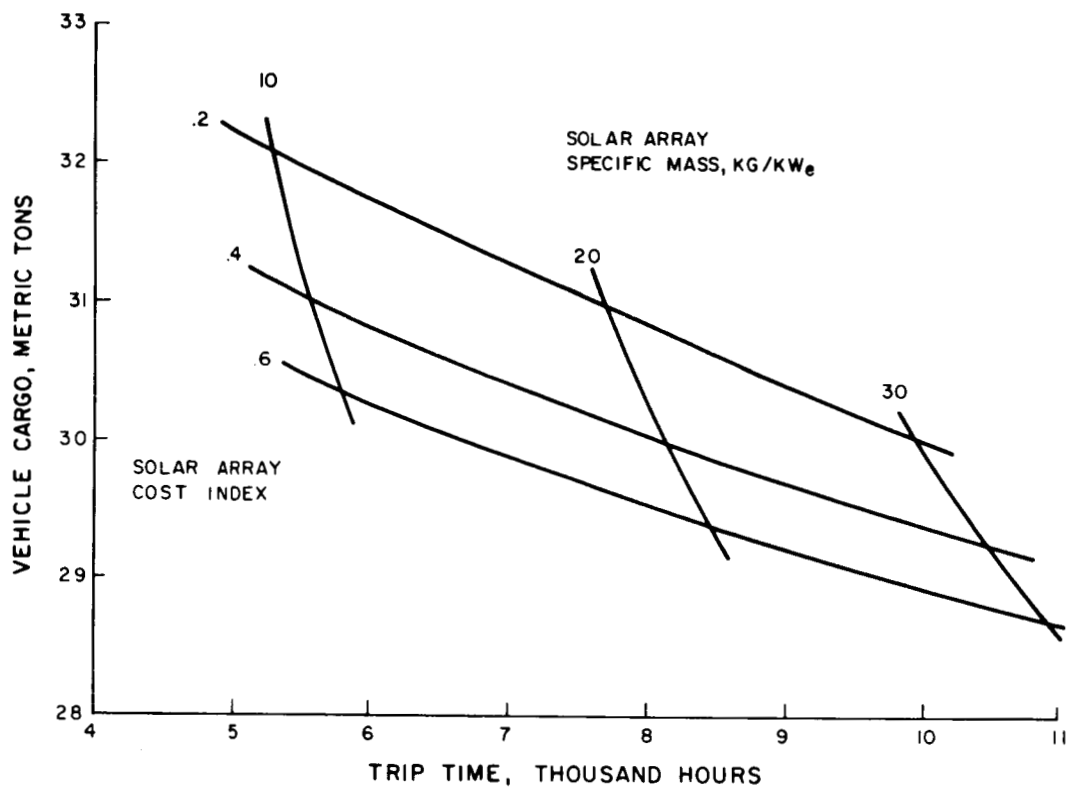


Figure 6-29. Vehicle Cargo and Trip Time Variation with Solar Array Cost Index and Specific Mass

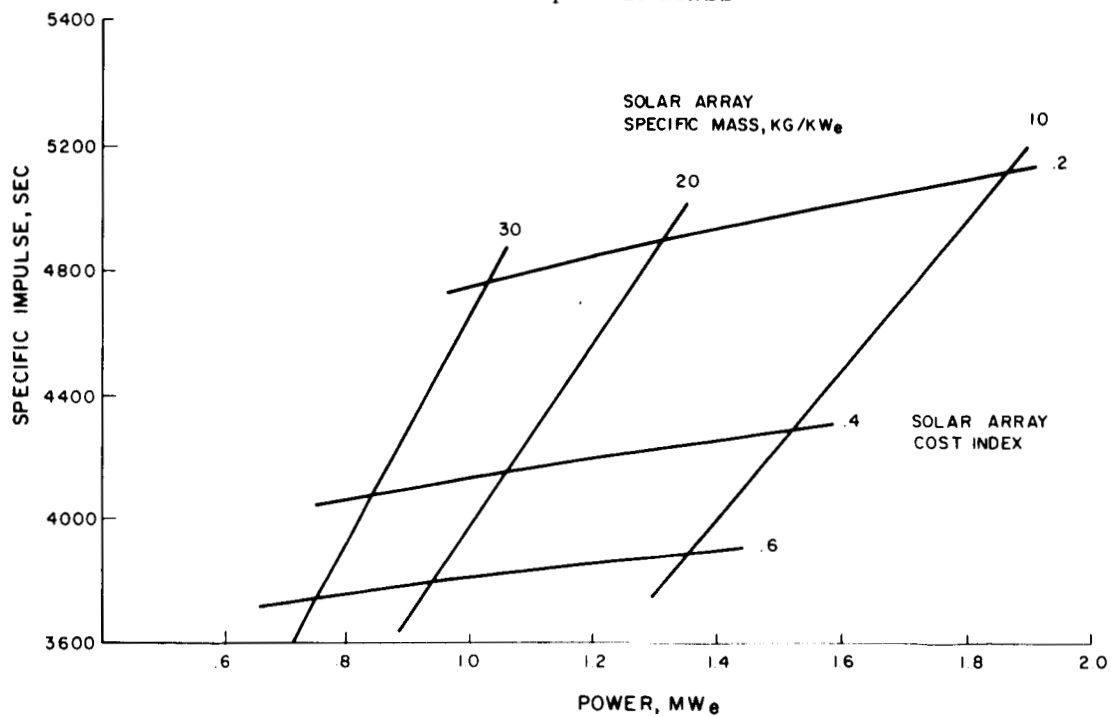


Figure 6-30. Specific Impulse and Power Variation with Solar Array Cost Index and Specific Mass

powerplant comparison. Thus, a lunar landing vehicle gross mass of 75 to 80 metric tons is approximately optimum for all electrical propulsion systems.

Figure 6-30 shows that the specific impulse ranges between 3800 and 5200 seconds; the power, between 0.7 and 1.9 MW_e. Assuming that 10 KG/KW_e can be attained, the solar array size is in the range of 1.4 to 1.9 MW_e.

6.4 COMPARISON OF NUCLEAR VERSUS SOLAR POWER

A comparison between nuclear and solar power sources for the electrically propelled lunar cargo operations involves many considerations in addition to performance. In the present study the performance potential was the only consideration explored in depth. Several of the other considerations have been identified and are discussed below in addition to the comparison of mission performance characteristics.

The performance comparison between nuclear and solar power systems is presented in Figure 6-31 on the basis of cost index and trip time. This comparison is based on

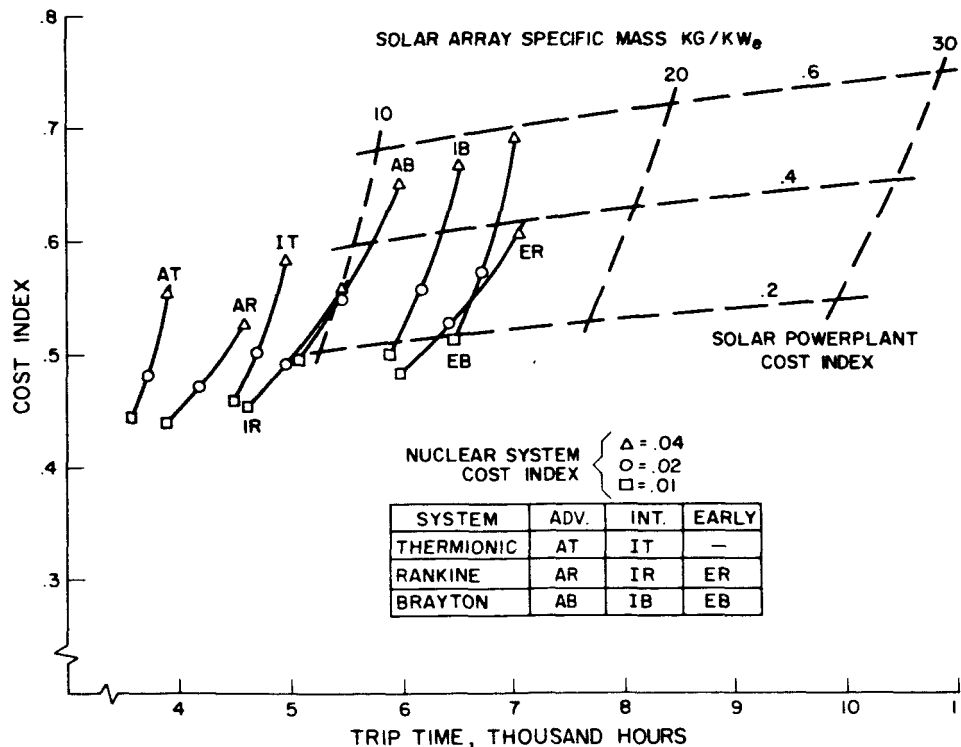


Figure 6-31. Performance Comparison Between Various Nuclear and Solar Power Systems

a 2000 ton cargo requirement, a 0.5 Time Worth Index, and a \$100 million booster cost. From this graph the advantage of many of the nuclear systems over solar power is readily apparent. The solar system at the potentially lowest specific mass of 10 KG/KW_e surpasses the early Rankine and all Brayton systems provided the Solar Power-plant Cost Index of 0.2 can be attained. This means a solar array constructed at \$200 per watt and operated for ten years or any combination thereof equivalent to \$20 per watt per year. An exact comparison between the various systems requires an improved cost analysis which is beyond the scope of the present study.

A comparison between nuclear and solar power systems should include several other considerations, which are related to system development, construction, operation and reliability. These are briefly mentioned below.

The nuclear powerplant requires an extensive development program. The required technology required to achieve the level of performance for electrical propulsion to be advantageous is not presently established. However, research programs are in progress, which show promise that the necessary technology can be achieved. Several more years of technology improvement are believed to be necessary before a nuclear powerplant concept can be selected for development, test and evaluation. The powerplant development goals, however, are less restrictive for the lunar cargo operation than for other space missions under study such as manned interplanetary. The lunar cargo vehicle travels unmanned, which relaxes the reliability requirement. Also, the mission time can be as short as 3 to 4 months, compared to 10,000 hours generally specified as a life qualification. On this basis of these relaxed requirements, a development period of five years seems possible, and the lunar cargo operations could commence in 1975.

The technology for the solar array at 10 KG/KW_e is also not established. Assuming that it can be achieved, several years are required to establish the technology, and more years for development, test and evaluation. Thus, even the solar system would likely not be available until 1975. On the basis of availability, the nuclear and solar power systems are at a standoff.

The nuclear powerplant is developed as an integrated system wherein the major cost is in the reactor assemble. The powerplant development program will require construction and test of several complete powerplant assemblies. Hence, its RDT&E cost will be many times the cost of production powerplant procurement, and this development cost is a significant influence on the cost effectiveness of the electrically propelled cargo vehicle operation. On the other hand, the solar system is an array comprised of perhaps 50 million silicon cells, each a generator of electrical power. The system development in this case can be based on a modularized approach and the development cost could be of the same or less magnitude than the cost of a complete operational assembly. Thus, the investment prior to a definite committment to operational use is minimal leading to a low risk program.

The modular characteristic of solar arrays also provides an excellent reliability situation. Cell and circuitry damage will tend to cause minor degradation in total power, which cause relatively small perturbations propulsion requirements and trajectory control by modest oversizing of the power supply. The nuclear system, conversely, can be subject to single failures, which bring about complete loss of power.

The solar array is large in dimensions. For one megawatt of electricity, 10,000 square meters of solar panels are required. This poses difficulties in spacecraft design, wherein folding structures must be utilized and ultra light structural support employed. The nuclear system is comparatively compact and can be launched in a fixed configuration requiring no deployment steps.

The solar array also requires pointing at the sun, whereas the nuclear system radiators can be oriented in any direction. The thrust vector of the electrical propulsion system, however, needs to be pointed parallel to the direction of motion. The angle between the sun and thrust vectors is continuously changing, which necessitates a rotating solar array assembly relative to the electrical propulsion system. This introduces mechanical complexity not required for nuclear systems.

Another complexity required for the solar power approach is the operational mode required for attractive performance. Whereas the nuclear system can be applied in a single-trip operational mode, the solar system requires a shuttle operation. Thus, the solar power approach demands earth orbit rendezvous and all the problem associated with this operation.

On the whole the lunar cargo operation favors the selection of the nuclear power approach. The developmental requirement can be simplified to enable early operation in a single-trip mode. The continuous improvement of performance can be applied to improve the cost, speed and flexibility of operation. The solar power approach could be considered as an interim step before nuclear power systems of advanced technology became available. Thus, the operational characteristics of electrical propulsion could be debugged using solar power, thereby avoiding the hazardous nuclear operation in the early stages. The lunar cargo operation provides an opportunity to develop increased life and reliability of nuclear power systems, which can later be used in manned interplanetary operations. This growth capability is another consideration favoring the nuclear power approach.

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APPENDIX
PRELIMINARY EXAMINATION OF THE PERFORMANCE
POTENTIAL OF A SOLAR POWERED LUNAR FERRY

PRELIMINARY EXAMINATION OF THE PERFORMANCE POTENTIAL OF A SOLAR POWERED LUNAR FERRY

SECTION A1

INTRODUCTION

This report supplements the previous nuclear-electric powered lunar ferry trajectory analysis (GE Document No. 65SD4361). Extensive performance and mission studies were conducted with an assumed nuclear system. The purpose of this report is to present the results of a preliminary examination of the performance potential of a solar powered lunar ferry and to make a comparison with that of a nuclear powered vehicle.

A lunar vehicle dependent upon solar power may be expected to encounter problems not normally associated with a nuclear powered system. That is, if it remains in the earth-moon plane, the vehicle will experience periods during which it is passing through the earth's shadow and will not be able to expose its solar panels to the sun. This constraint considerably complicates the trajectory analysis.

In order to analyze solar powered operation, three distinct types of trajectories were considered which were compatible with the shadowing effect normally encountered during a lunar transfer.

A trajectory profile identical to that of a nuclear powered transfer can be maintained by utilizing stored electrical energy during each shadow period. A powerplant weight penalty may be expected with this approach due to the weight of the energy storage system and the additional solar cells required to feed it.

A second option considered was a "three dimensional" type of trajectory which avoids shadow completely by spiraling out from the earth at an angle sufficiently inclined to

the ecliptic. The lunar approach trajectory has to be similarly inclined and matching the two trajectories in different planes will require an increase in propulsion requirements.

The third approach considered was a pulsed type of operation whereby the vehicle uses propulsion only during periods of sunlight and coasts through the shadow regions. This type of trajectory will obviously result in longer trip times and may lead to slightly larger propulsion requirements.

Combinations of these three modes of operation may be of interest but were considered beyond the scope of this preliminary study.

SECTION A2

SUMMARY

Of the three modes of solar operation considered, the one that appeared the most attractive when compared with the nuclear system was the pulsed propulsion mode.

With energy storage to be used during darkness or three dimensional shadow free propulsion, powerplant weight penalties ranging from 60 to 100 percent of that of a nuclear system will be incurred. In addition, a three dimensional trajectory encounters matching problems for the lunar approach orbit.

Pulsed propulsion, on the other hand, yielded only moderate propulsion penalties and trip time penalties ranging from 16 to 20 percent.

A solar powered vehicle with pulsed propulsion capability would be attractive for a multi-trip lunar ferry operation, whereas previous studies have indicated that the single trip mode may be more suitable for a nuclear system. Once launched, a solar powered vehicle becomes, in a sense, a resource in space, and its cost can be distributed over a time period involving several missions.

Due to the earlier availability of solar arrays with the required power capabilities, they would provide a logical transition to the nuclear-electric system for lunar logistics operations.

SECTION A3

ANALYSIS

As mentioned in the Introduction, trajectory analysis of a solar powered lunar ferry presents unique problems not associated with a nuclear powered vehicle. Figure A-1 illustrates a conventional earth departure spiral trajectory for a lunar transfer. It will lie within the earth-moon plane which is inclined at about five degrees to the ecliptic; consequently the vehicle will pass through the earth's shadow during each revolution and the solar panels will not be able to function. The methods used to investigate the potential performance capabilities of the three solar powered modes considered are discussed in this section. They are presented in the order of the difficulty encountered in the calculation procedure due to the earth shadowing effect. In all cases, the inclination of the moon's orbit to the ecliptic is neglected and it is assumed that low thrust propulsion is initiated from a 483 km (300 mi) circular orbit with a specific impulse of 5000 seconds.

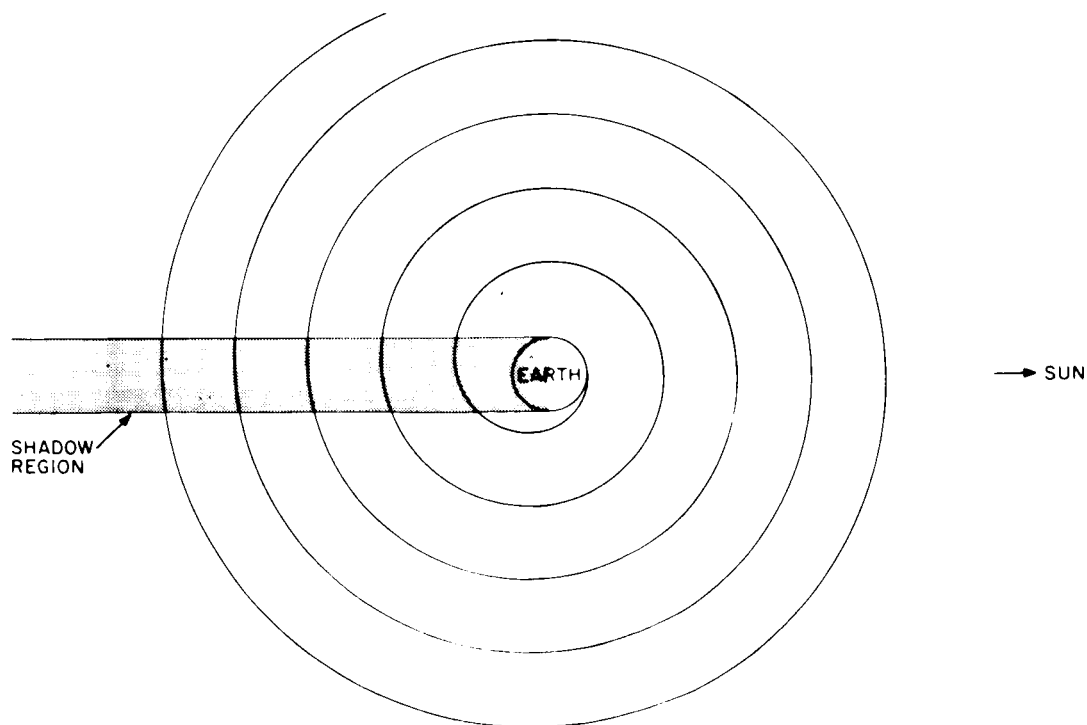


Figure A-1. Two-Dimensional Earth Departure for Lunar Transfer

A3.1 ENERGY STORAGE DURING SUNLIGHT

Using the propulsion characteristics of a typical optimum nuclear system, the relationship between time in shadow and in sunlight was examined. This should be a direct indication of the powerplant weight penalty incurred due to the energy storage requirement. Figure A-2 indicates the shadow time that may be expected during earth departure for a typical lunar transfer.

The left hand curve shows the ratio of time per revolution to time in sunlight as a function of the orbit parameter, P (semi-latus rectum). The power generation capacity of the vehicle must be sized based on the initial orbital altitude. For the 483 km initial altitude considered in these investigations ($P = 6854$ km), Figure A-2 indicates that energy storage

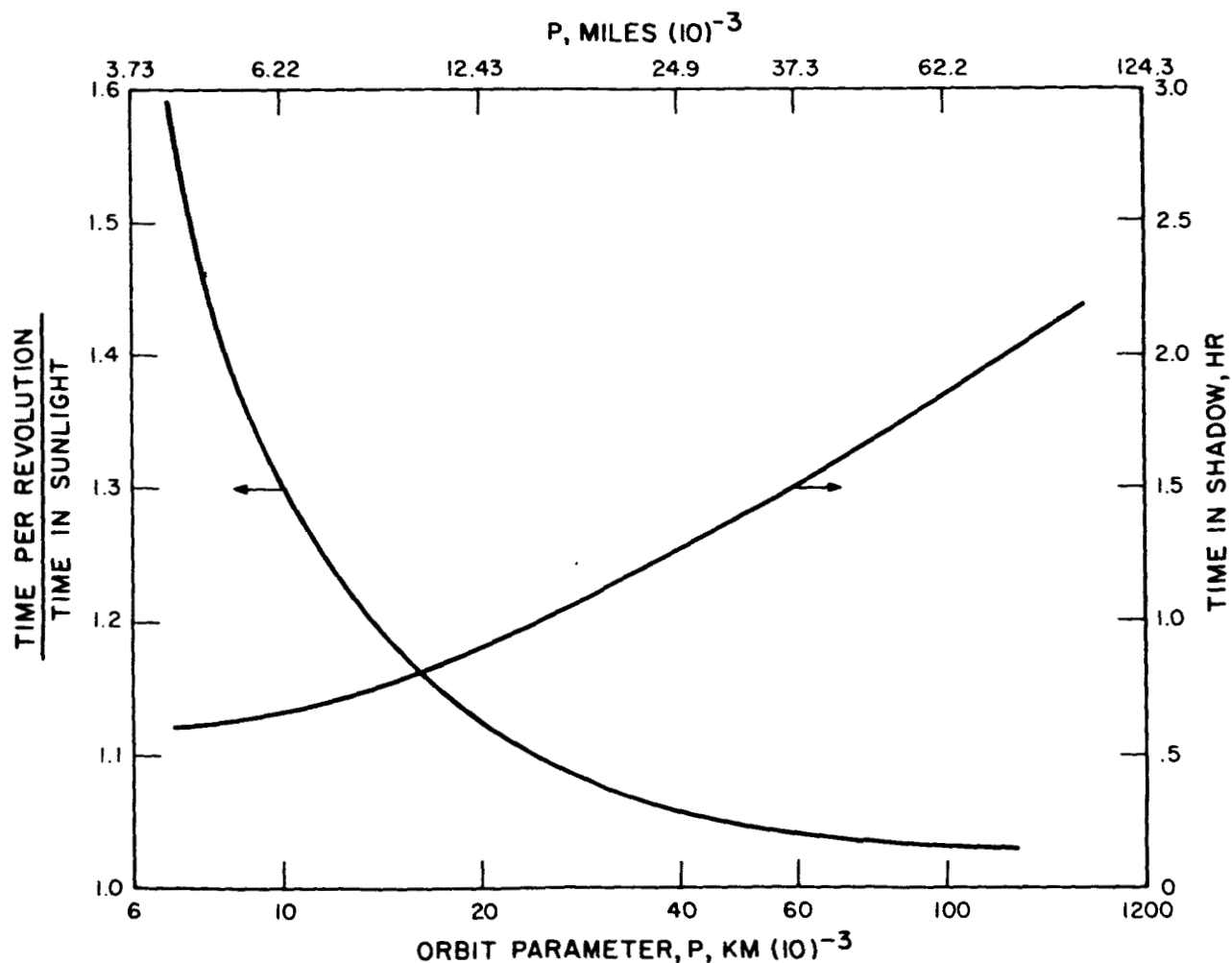


Figure A-2. Energy Storage Requirements

will require a 59 percent larger power generating capacity than a comparable nuclear system. The right hand curve shows the variation of the time in shadow with orbit size. The most severe energy storage requirement will be of the order of 2 to 2.5 hours at the end of earth departure.

As a result of these two considerations, the total effect of the excess power generation and associated energy storage equipment can be expected to result in a powerplant weight penalty in excess of 60 percent. It has been concluded, therefore, that the energy storage mode of solar operation is not competitive with a comparable nuclear system.

A3.2 THREE-DIMENSIONAL SHADOW FREE TRAJECTORY

The lunar orbit is inclined at about five degrees to the ecliptic. If a thrust pattern is employed such that the vehicle leaves this plane with a high inclination angle, the shadow region may be avoided entirely up to a point. To attain a lunar rendezvous, a similar highly inclined lunar approach trajectory will be required.

A3.2.1 Orbital Geometry

Figure A-3 shows the departure geometry assumed with respect to the ecliptic plane, neglecting the inclination of the moon's orbit. The hour angle of the initial orbit δ_o , is dependent upon the inclination and initial altitude according to the following relationship:

$$\delta_o = \arcsin \frac{R_s}{(R_s + h) \sin I} \quad (A1)$$

where,

R_s is the radius of the earth's surface = 6372 km

h is the assumed initial orbital altitude = 483 km

I is the orbit inclination angle.

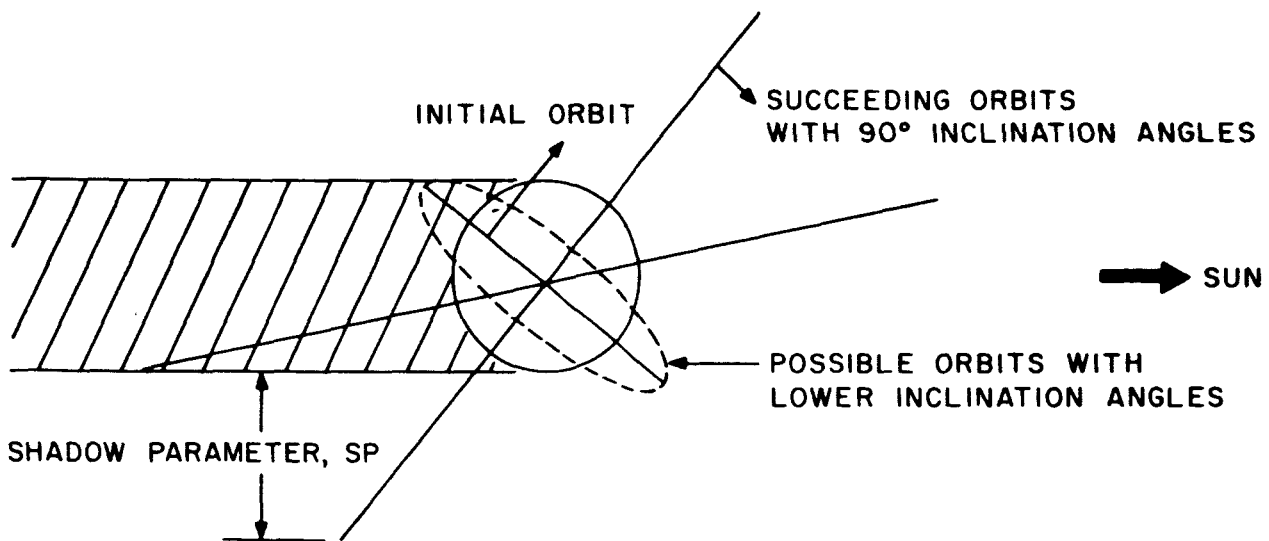


Figure A-3. Three-Dimensional Departure Geometry

Initially it is assumed that the vehicle is just leaving the shadow region in order to maximize the time in sunlight. For the assumed initial conditions the inclination angle is restricted by equation A1 to a range of 68.5 to 90 degrees. As propulsion progresses during earth departure, the size of the orbit continues to grow, and the shadow region precesses with respect to the fixed orbital plane at a rate, $\dot{\psi}$, of .0405 degrees per hour. This characteristic is indicated in Figure A-3 by a precessing orbit plane with respect to a fixed shadow region. The vehicle will eventually re-enter a region of darkness when the projection of its radius, R , on the earth-sun line becomes less than the radius of the earth's surface, or when,

$$S. P. = R \sin (\delta_0 + \dot{\psi} t) \sin I - R_s < 0, \quad (A2)$$

which defines the shadow parameter shown in Figure A-3.

Eventually, the shadow parameter will approach zero. The time at which it vanishes will be dependent upon the inclination and the rate of increase in the size of the orbit which in turn depends upon the thrust to weight ratio. This available propulsion time increases from the minimum 68.5 degree orbit to the maximum 90 degree orbit.

A3.2.2 Shadow Free Departure Limitations

In order to determine the feasibility of maintaining propulsion during sunlight, with a three-dimensional mode of operation, portions of the trajectory analysis of the previous Lunar Ferry Study were utilized. In this analysis the following differential equation for the rate of change of the orbit parameter, P, was developed:

$$\frac{dP}{dt} = \frac{2g (T/W_o) P \sqrt{P/GM}}{1 + e \cos \phi} \quad (A3)$$

where,

g is the sea level gravitational constant = 127140 km/hr²

T/W_o is the thrust to weight ratio

GM is the universal gravitational constant = 5.1648(10)¹² km³/hr²
(for the earth)

e is the orbit eccentricity

φ is the true anomaly

In addition, it has been found that a cubic polynomial in eccentricity as a function of true anomaly may be expressed as,

$$e^3 + \frac{14 \cos^2 \phi - 4}{\cos \phi (3 \cos^2 \phi - 1)} e^2 + \frac{6}{\cos^2 \phi} e + \frac{4}{\cos \phi (3 \cos^2 \phi - 1)} = 0 \quad (A4)$$

Figure A-4 shows this relationship in the region of interest for the lunar ferry mission.

By assuming a transverse thrust, T, and combining equation A4 with the parameter,

$$Z = \frac{P^2 T}{GM} = \frac{e (1 + e \cos \phi)^2}{\sin \phi (2 + e \cos \phi)} \quad (A5)$$

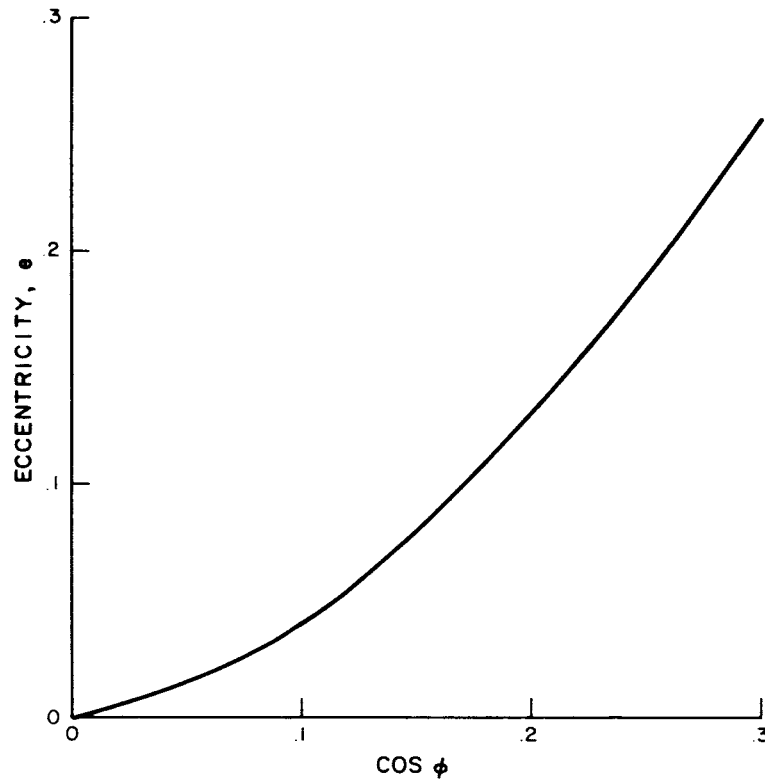


Figure A-4. Eccentricity as a Function of True Anomaly

the relationship shown in Figure A-5 can be developed. This, in turn, can be approximated quite closely by the following quadratic:

$$e \cos \phi = .091667Z + 3.50 Z^2 \quad (A6)$$

By substituting equation A6 into equation A3 and using the definition of Z, equation A3 can be integrated and the following expression obtained relating time, thrust to weight ratio (acceleration, T), and the orbit parameter, P:

$$t = \frac{V_c}{T} (1 - .030556Z + .50 Z^2) \bigg|_{P_o}^P \quad (A7)$$

where,

V_c is the circular velocity, $\sqrt{GM/P}$

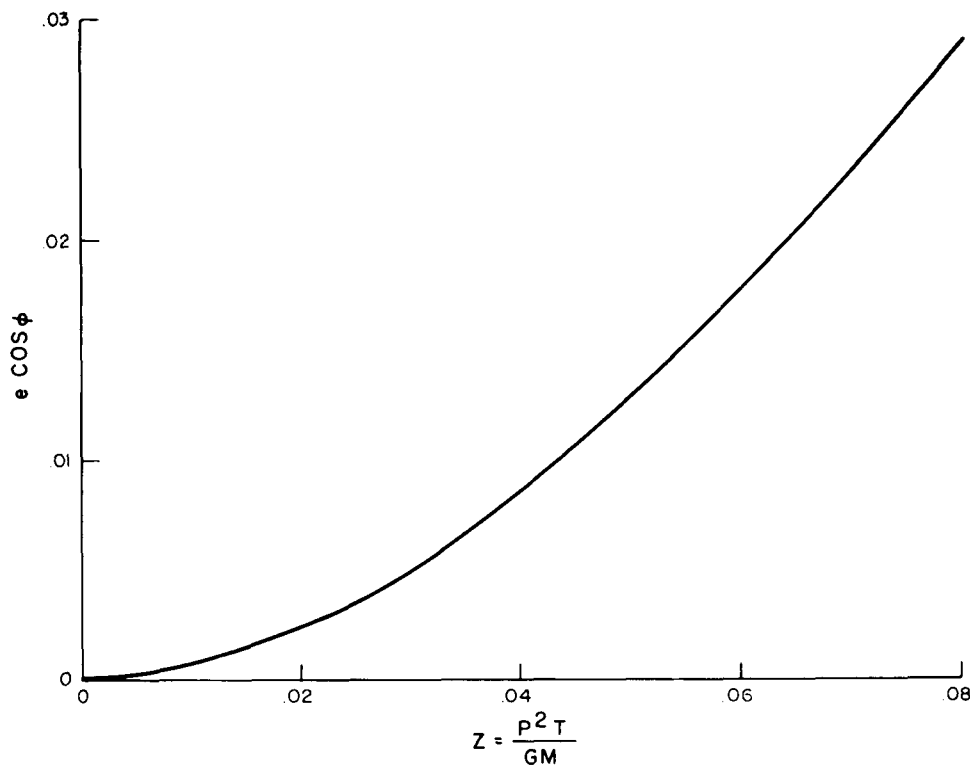


Figure A-5. Orbital Parameter Relationship

Thus, for a given inclination angle and thrust to weight ratio, the variation of the shadow parameter as defined by equation A2 can be determined. This characteristic is illustrated in Figure A-6 for several thrust to weight ratios with a 90 degree inclination. As might be expected, higher thrust to weight ratios yield higher values of P , and hence larger orbits before the shadow parameter becomes negative. The locus of points where the shadow parameter goes to zero represents the maximum attainable values of P with continuous sunlight propulsion.

These maximum values are shown in Figure A-7 as a function of thrust to weight ratio for the two extreme inclinations. Superimposed are the minimum earth departure requirements to reach the lunar sphere of influence in the earth-moon plane. Note that the earth departure thrust to weight ratio must be at least $6.75 (10)^{-5}$ for a 90 degree inclination and $8 (10)^{-5}$ for 68.5 degrees.

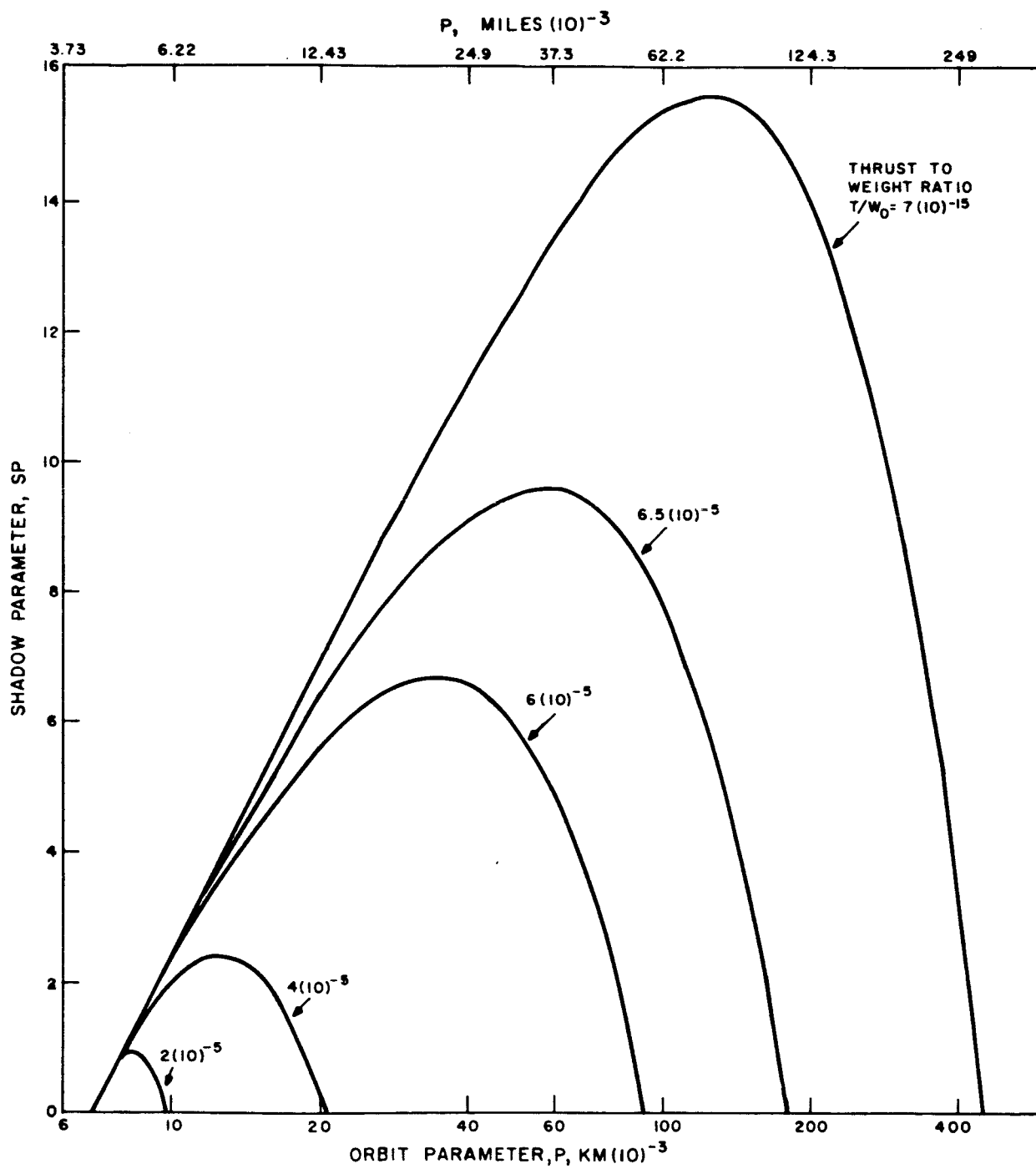


Figure A-6. Three-Dimensional Shadow Free Departure Characteristics

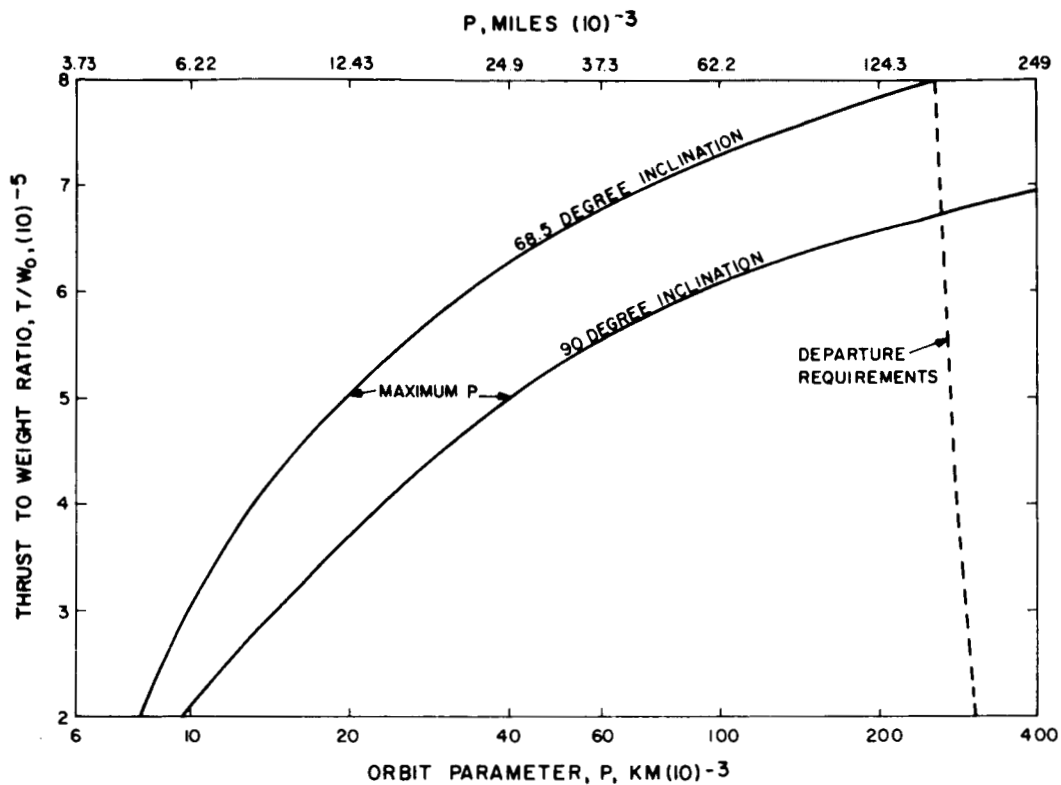


Figure A-7. Minimum Thrust to Weight Ratio Requirements

Since the nuclear-electric lunar ferry studies indicated optimum thrust to weight ratios of the order of $4 (10)^{-5}$, it would appear that with a three-dimensional, solar powered earth departure, there is a potential powerplant weight penalty of 70 to 100 percent.

A3.2.3 Lunar Approach Characteristics

The lunar approach orbit characteristics for both the conventional nuclear-electric two-dimensional transfer and the solar-electric three-dimensional transfer are shown in Figure A-8. The trajectory characteristics are shown in the earth-moon plane and the velocity diagrams are shown at the earth-moon transition point (the lunar sphere of influence) and in the plane perpendicular to the earth-moon line. In the two-dimensional case, the vehicle velocity and the lunar velocity relative to the earth are co-linear and their difference, the vehicle velocity relative to that of the moon is of the order of 20 percent of the lunar velocity. The resulting lunar approach orbit is highly elliptical.

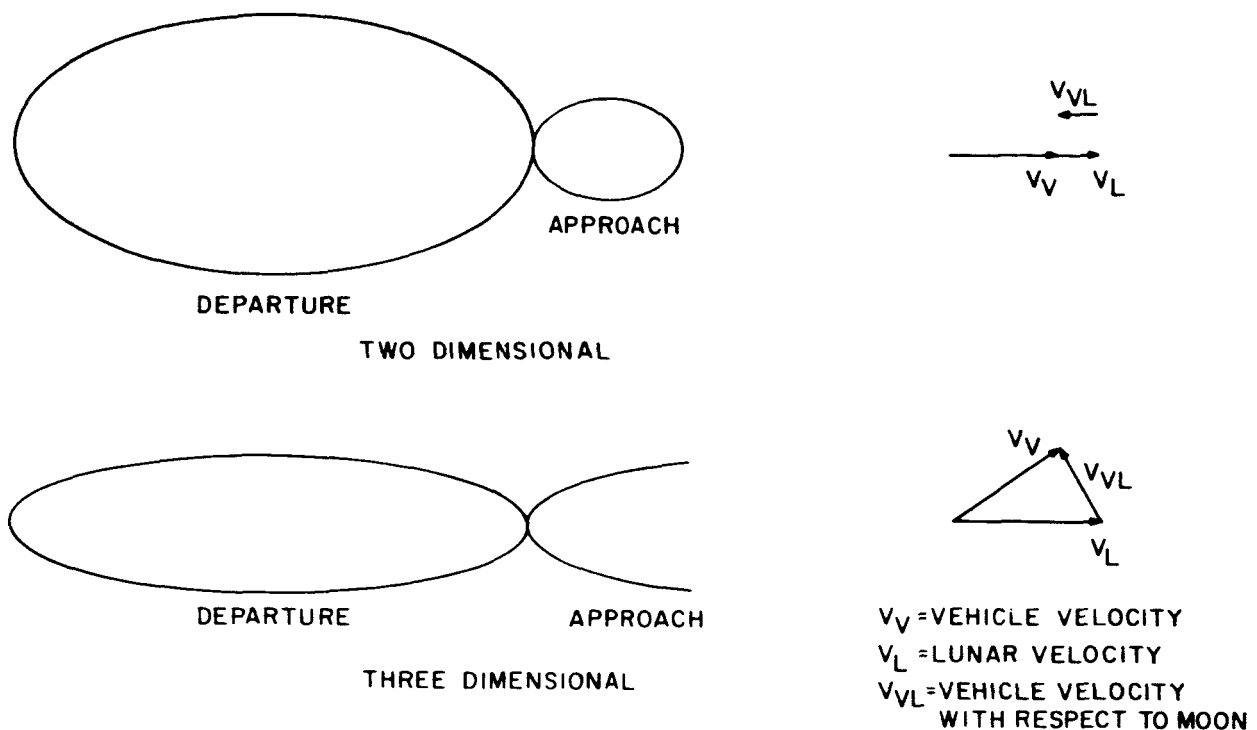


Figure A-8. Lunar Approach Characteristics

Conversely, the three-dimensional vehicle velocity is not co-linear with the moon's and the resulting vehicle velocity relative to the moon is substantially higher. This results in a hyperbolic lunar approach orbit with an eccentricity of the order of 10 to 15. It is extremely doubtful that a low thrust lunar capture can be achieved before the vehicle leaves the lunar sphere of influence permanently. The lunar capture problem can be reduced, but at the expense of higher thrust-weight ratios or more propulsion penalty for the earth departure mode.

As a result of these considerations and the departure penalties pointed out in the preceding section, it has been concluded that the three-dimensional, shadow free mode of operation is unlikely to be attractive.

A3.3 PULSED PROPULSION

The discontinuities associated with a pulsed mode of operation make it the most difficult to analyze of the three types of trajectories considered, but it may also be the most promising. It was found necessary to develop a new computer program to handle this unique type of lunar transfer.

A3.3.1 Trajectory Characteristics

Figure A-9 compares the variation of trajectory characteristics between a continuous propulsion transfer and one with a coast period. With continuous propulsion, the line of apsides becomes synchronized with the vehicle after an initial period, and the vehicle stabilizes out at a position less than 90 degrees from perigee. With the discontinuous pulsed mode, however, the line of apsides at first tends to align itself with the center of the shadow region at apogee; there is a subsequent gradual precession of the line of apsides with respect to the shadow region as the region shrinks with increasing orbit altitude. This institutes an oscillatory trend in some of the trajectory characteristics and results in complications in the calculation procedures.

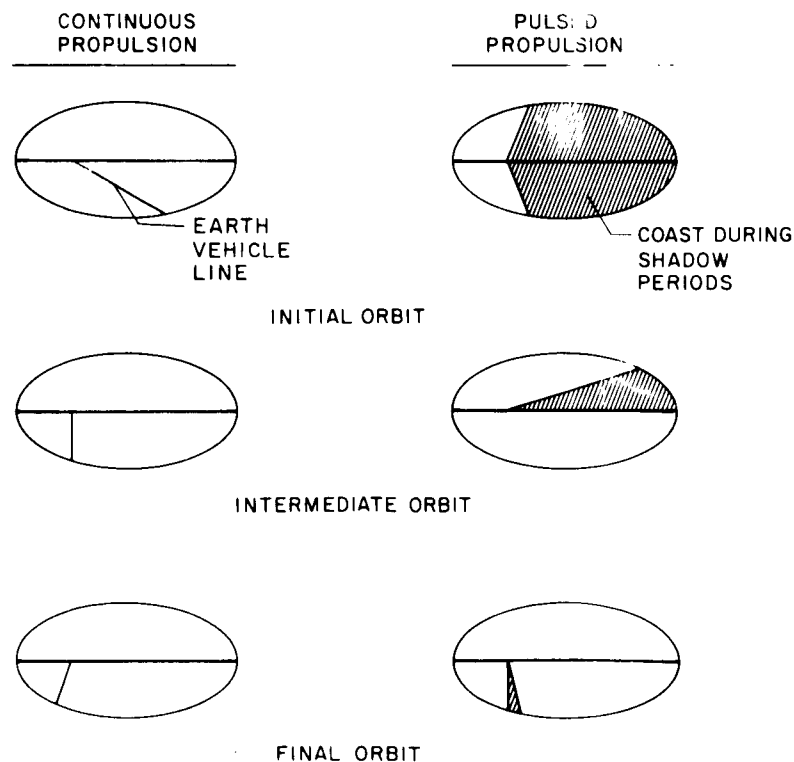


Figure A-9. Pulsed Propulsion Trajectory Characteristics

For example, since orbit eccentricity increases only within ± 90 degrees of perigee, it might be expected that the eccentricity with pulsed propulsion would increase during the initial orbits to a maximum, and then decrease during intermediate orbits to a minimum before increasing to earth escape. Whereas, with continuous propulsion, one would expect a continuous increase in eccentricity throughout the trajectory.

A3.3.2 Calculation Procedure

Due to the nature of the pulsed propulsion type of trajectory, it was found necessary to employ a simultaneous numerical integration of the differential equations of motion. A fourth order Runge-Kutta method was used in conjunction with the following four equations:

$$\frac{dR}{dt} = U \quad (A8)$$

$$\frac{dU}{dt} = \left[\left(\frac{H}{R} \right)^2 - \frac{GM}{R} \right] R \quad (A9)$$

$$\frac{dH}{dt} = aR = a_0 R \left(1 - \frac{a_0 t}{V_j} \right) \quad (A10)$$

$$\frac{d\theta}{dt} = \frac{H}{R^2} \quad (A11)$$

where,

U is the radial velocity

H is the angular momentum

a is the acceleration

V_j is the jet velocity

θ is the vehicle angle measured from an initial reference at $t = 0$.

other symbols have been previously defined.

(Equation A9 assumes there is no radial thrust component)

In addition, near the end of each propulsion period, a quadratic extrapolation procedure was used to predict the point at which the precessing shadow region would be re-entered.

A3.3.3 Eccentricity Variation and Propulsion Requirements

As a result of the unusual characteristics of a pulsed mode of operation, the orbit eccentricity was found to exhibit a pattern similar to that shown in Figure A-10. For these particular conditions, the eccentricity of the pulsed trajectory hits a maximum of 16 percent, then decreases to a minimum of 8.5 percent and then increases to ultimately achieve earth escape. Also shown is the steadily increasing eccentricity with continuous propulsion. It is significant that there is not much difference in propulsion time between the two cases until high orbit parameters are reached.

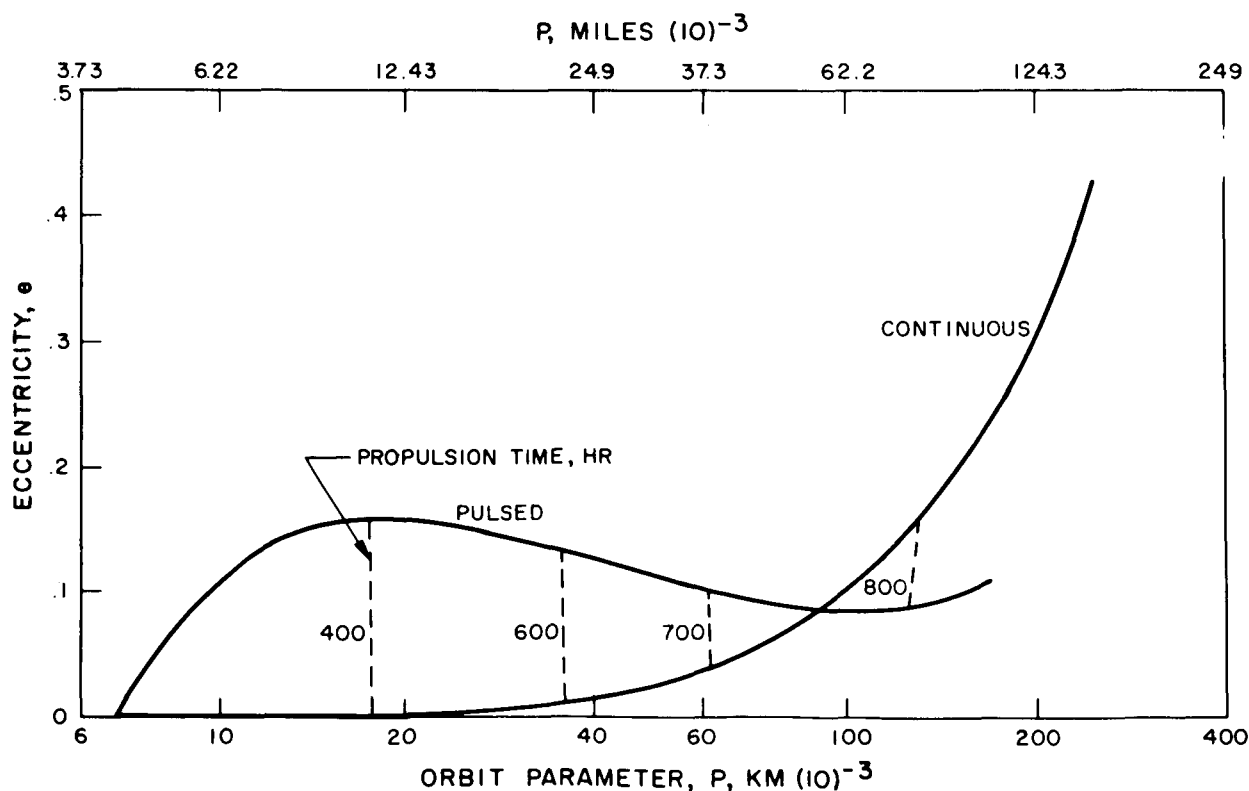


Figure A-10. Pulsed Propulsion Orbit Characteristics

Figure A-11 illustrates the difference between the characteristic velocity requirements for the pulsed mode of operation and the comparable continuous mode. Note that the difference is about 3.5 percent at low altitudes and that it decreases to about 2 percent for the earth departure orbits required for earth-moon transfer. A comparable penalty can be expected for the pulsed lunar capture propulsion. The small difference between the two modes of operation indicates that the results of the prior lunar ferry parametric performance studies can be used as a first approximation for solar-electric vehicle performance. The trip time requirements must, however, be modified to account for the coasting time during shadow.

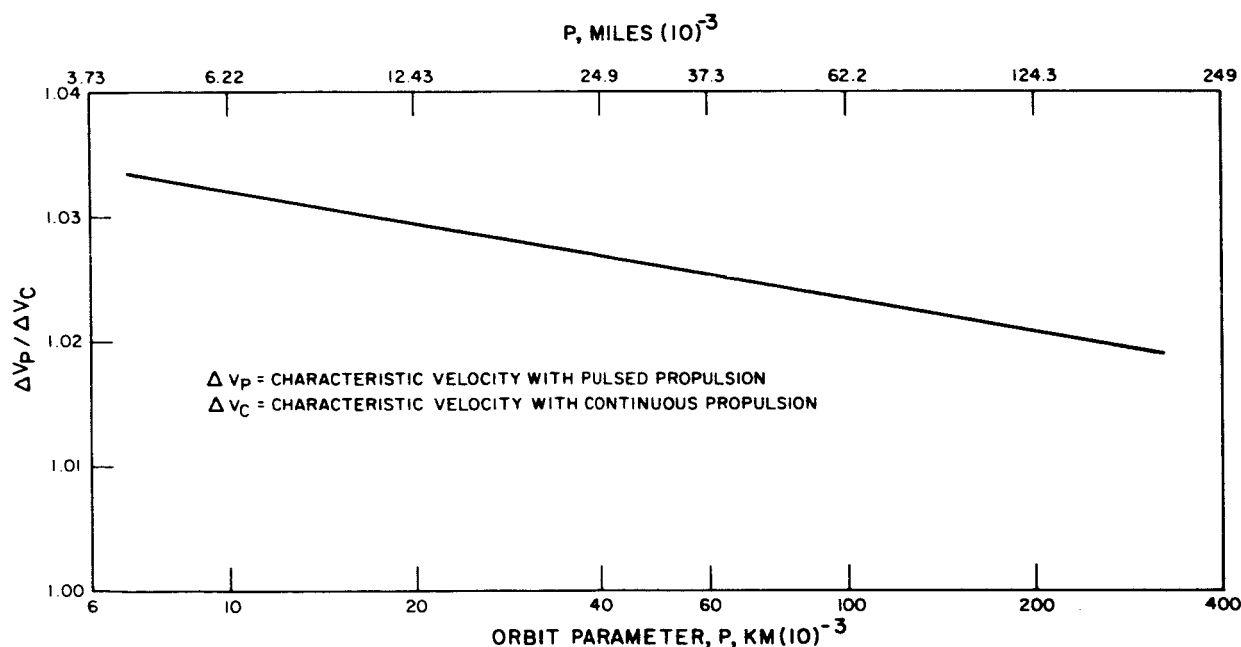


Figure A-11. Characteristic Velocity Comparison

A3.3.4 Trip Time Penalty

The relationship between trip time, propulsion time and thrust to weight ratio for trajectories with a coast during periods of darkness is summarized in Figures A-12 to A-14. Superimposed on the first two figures are the earth departure requirements

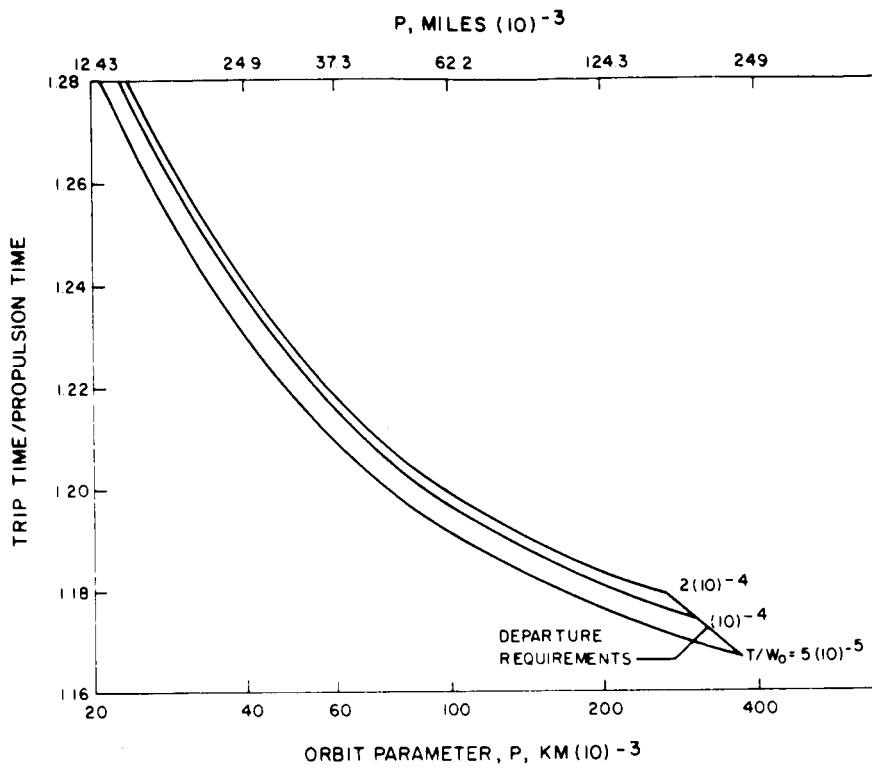


Figure A-12. Earth Departure Trip Time Penalty with Pulsed Propulsion

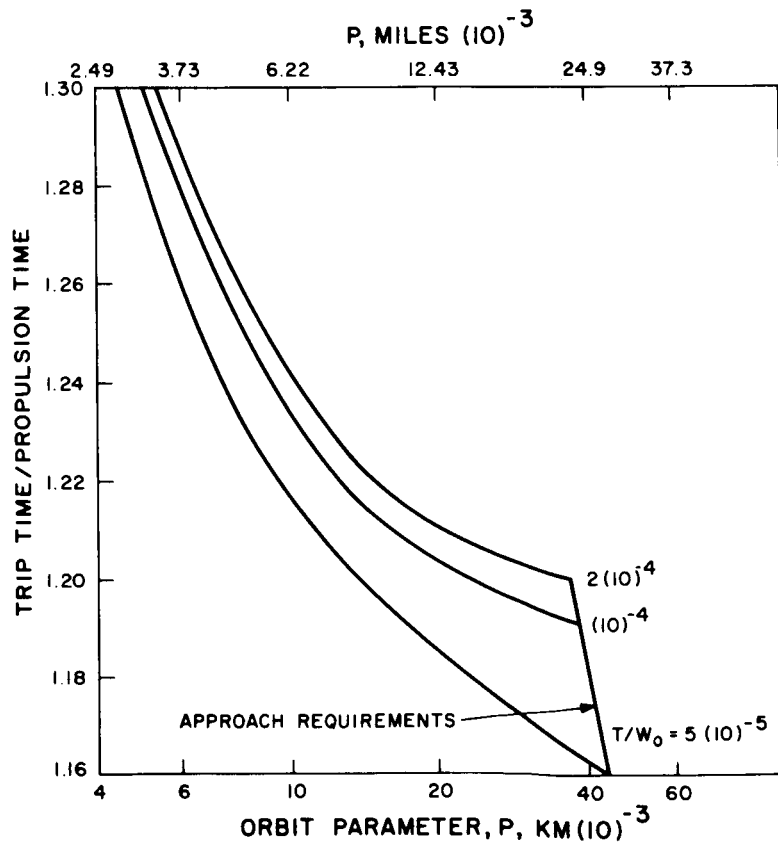


Figure A-13. Lunar Approach Trip Time Penalty with Pulsed Propulsion

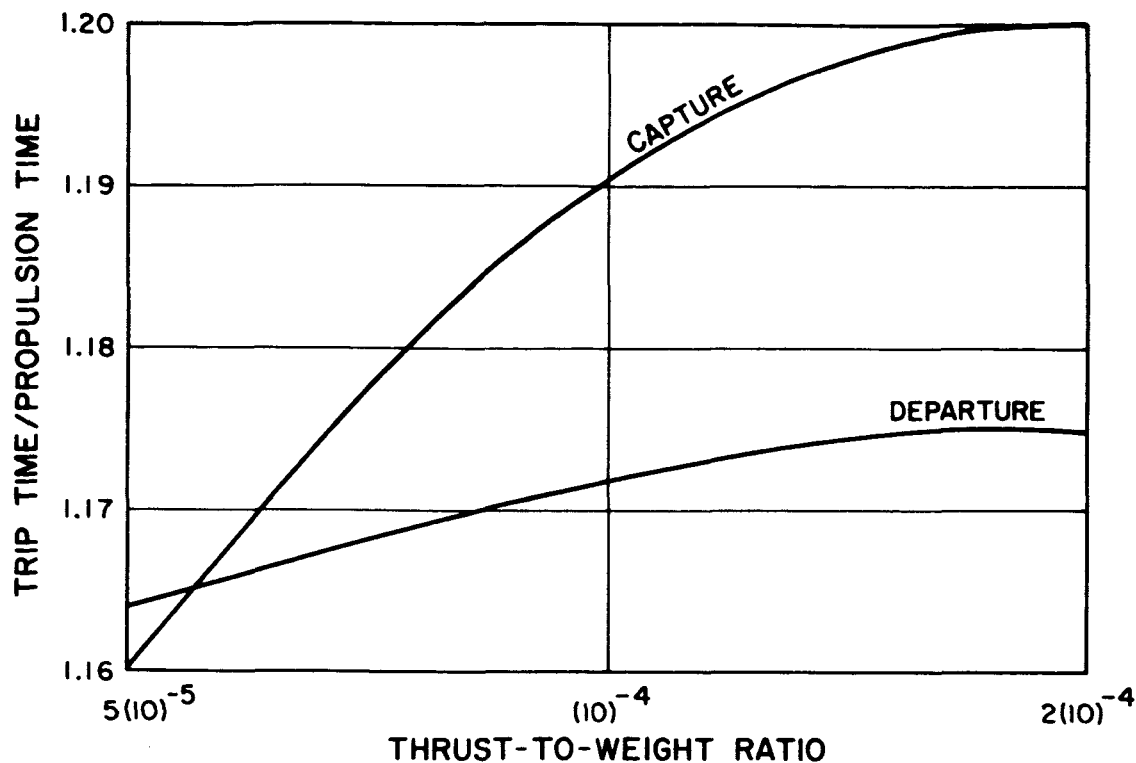


Figure A-14. Trip Time Penalty

(propulsion cutoff) and the lunar approach requirements (propulsion initiation). The earth departure penalties are of the order of 17 to 18 percent and the lunar approach penalties of the order of 16 to 20 percent. Figure A-14 summarizes the trip time penalties for both the earth departure and lunar approach trajectories as a function of thrust to weight ratio. Since the lunar approach propulsion times generally run about 20 percent of the earth departure times, the overall trip-time penalty can be expected to be quite similar to the earth departure penalty.

For a nuclear powered system, previous cost analysis studies indicated that due to its relatively limited life (5000 to 10,000 hr), it would be most advantageous to use it for a single trip type of lunar ferry operation. Since the solar powered system can apparently deliver approximately the same payload with a maximum increase in trip time of 20 percent, its much longer lifetime (five to ten years) would suggest that it be used for multi-trip missions. Thus, the probable higher cost per unit power for solar panels compared to a nuclear source would diminish considerably as the number of missions increased.

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